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CONTENTS

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AN AIR-BREATHING SATELLITE BOOSTER

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Vol. 7, No. 6: Page 235, June 1961

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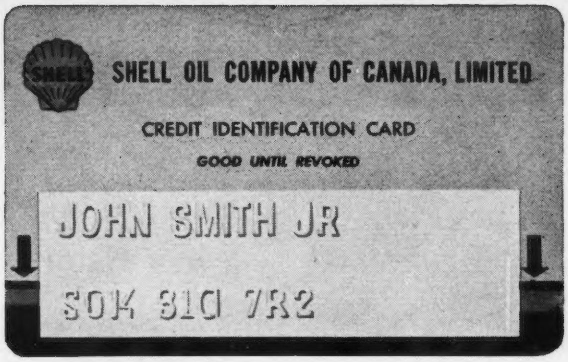

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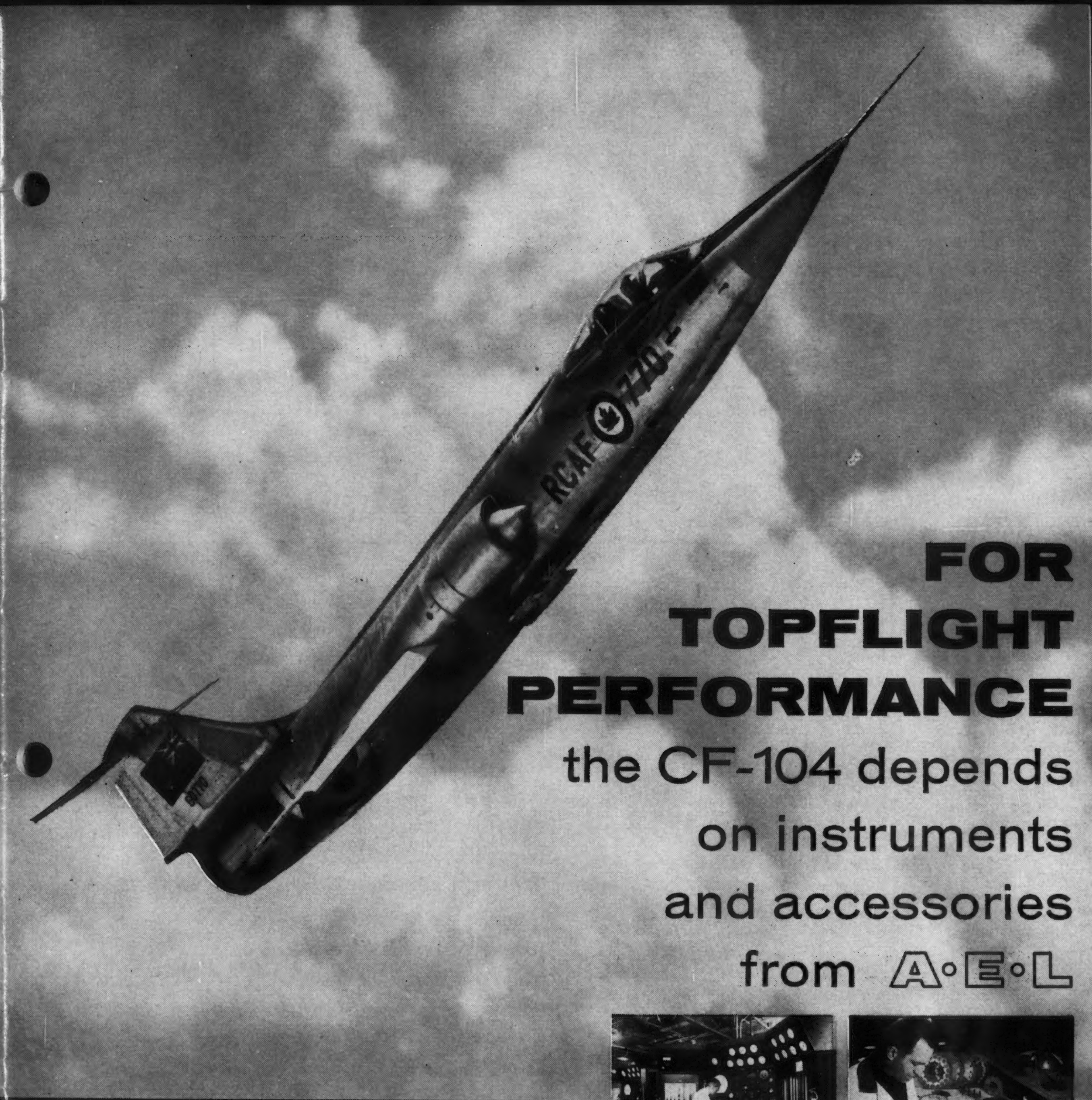

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CANADIAN AERONAUTICAL INSTITUTE
JOURNAL

AN AIR-BREATHING SATELLITE BOOSTER†

by Prof. J. H. T. Wu,* M.C.A.I. and S. Molder**

McGill University

SUMMARY

A three stage, composite, earth satellite launching system has been studied. The first stage provides power for takeoff, initial climb and acceleration to the second stage ramjet starting velocity. Two trajectories of the second stage (a recoverable, ramjet powered lifting vehicle) have been computed on an IBM 650 digital computer. The initial conditions of the two trajectories are the same except that for one the L/D ratio is 6.0 and for the other it is 3.0. The third stage is a rocket which takes its payload to orbital conditions.

The work of this report is devoted mainly to the performance of the second (ramjet) stage. Ramjet performance analysis brings out an important parameter quite similar to wing loading; it is here called the intake loading and is found by dividing the weight of the ramjet vehicle by the intake area, W/A. It is shown that W/A is a scaling factor for ramjet performance.

The calculations show that it is possible to boost into orbit a payload which is 3.25% of the initial all-up weight.

INTRODUCTION

Up until the present time all known space launchings have been accomplished with multistage ballistic rockets. These rockets have carried very large fuel and oxidant loads and consequently very small payloads.

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†Paper read at the Joint I.A.S./C.A.I. Meeting in Montreal on the 18th October, 1960.

*Associate Professor, Hypersonic Group

**Research Associate, Hypersonic Group

As the demand for larger orbiting and space exploration vehicles increases, consideration should be given to the boosting capabilities of air breathing propulsion devices. The main advantages, when compared with rocket boosting systems, are:

(1) oxidant is supplied by the atmosphere thereby eliminating a large oxidant weight penalty,

(2) aerodynamic pressure is available for support if the vehicle is provided with lifting surfaces. This decreases the thrust requirement, since only a very small amount of thrust is used for support, the rest being used for acceleration, and

(3) recovery of expended stages may be effected by a glide descent and landing, enabling these stages to be re-used.

The problems associated with an air breathing booster vehicle are mainly those of development. The air intake for an accelerating ramjet, for example, must perform efficiently over a wide velocity range. The achievement of efficient supersonic combustion would eliminate the necessity for decelerating the hypersonic free stream to subsonic velocities before combustion. This would carry with it considerable gains in overall efficiency and at the same time would decrease internal temperatures and pressures. External combustion techniques may alleviate both of these problems.

Hydrogen, although it gives a high specific impulse when burned with air, presents containment problems because of its low density even in its liquid state.

LIST OF SYMBOLS

A	ramjet air intake area, ft^2
a	acceleration, ft/sec^2
$b_{1,2}$	coefficients in the equation for I_{sp}
C	L/D ratio
$c_{1,2,3}$	coefficients in the equation for sfc
D	drag, lb
g	acceleration due to gravity, ft/sec^2
h	altitude, ft
I_{sp}	air or fuel specific impulse, sec
L	lift, lb
M	Mach number
p	payload fraction of any stage
r	radial distance measured from geocentre, ft
sfc	specific fuel consumption, $\dfrac{\text{lb/sec}}{\text{lb}}$
s	dead weight fraction
T	temperature, °R
$\mathcal{T}$	thrust, lb
t	time, sec
V	velocity with respect to a stationary earth, ft/sec
W	weight, lb
α	angle between thrust and velocity vectors, rad
ϕ	angle of climb with respect to local horizon, rad
ψ	angle of thrust with respect to initial horizon, rad
ρ	density of atmosphere, slug/ft^3

Subscripts

f	burn-out conditions
i	initial conditions
o	sea level

Superscripts

$\bar{}$	mean value
$'$	differentiation with respect to V
$\cdot$	differentiation with respect to t

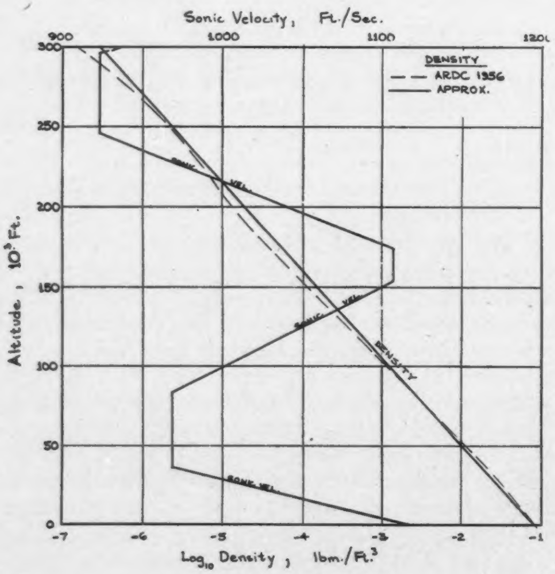

Figure 1

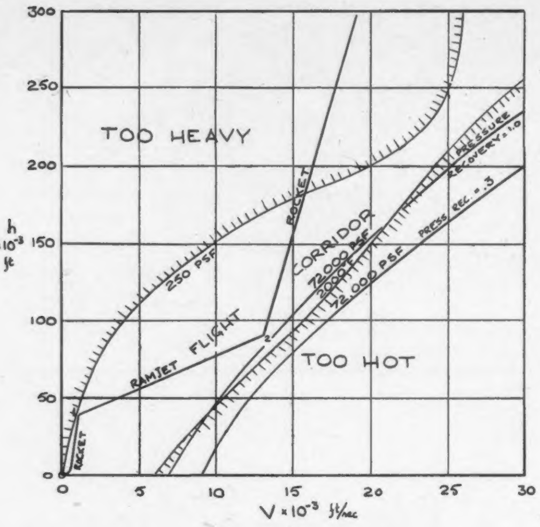

Figure 2
The flight corridor

THE ATMOSPHERE AND SOME PRACTICAL LIMITATIONS

The temperature and density variations of the ARDC Model Atmosphere (1956)[1] are shown in Figure 1 by the dashed lines. The solid lines indicate the values approximated for simplified computations.

Figure 2 is the well known flight corridor. For sustained flight at any altitude and speed, the vehicle must not exceed equilibrium skin and stagnation temperatures bearable by current high temperature materials (2000°F), this precludes prolonged flight in the region in Figure 2 marked "Too Hot". On the other hand, lifting stages must remain in a region of the h-V diagram where sufficient dynamic pressure is available for support and thus must not enter the region marked "Too Heavy". In addition to the high temperature and the low dynamic pressure limitation there exists a limit to the internal pressure in the engine, which cannot be exceeded for reasons of engine strength to weight consideration. Reference 2 gives this upper limit as 72,000 psf. The location of this limit in the flight corridor depends on the total pressure recovery of the intake and on the Mach number after diffusion. The lines shown in Figure 2 are for total pressure recoveries of 1.0 and 0.3, and for a Mach number at the diffuser exit equal to 4.0. Thus continuous flight of a ramjet powered vehicle is limited to the region marked "Flight Corridor".

THE LAUNCHING SYSTEM

The launching system will consist of three stages. The first stage is a jettisonable rocket cluster. Its purpose is to provide thrust for takeoff and for initial climb and acceleration. The takeoff is of shallow climb angle, airplane type. A 10,000 ft jet runway is probably suitable. After takeoff the vehicle climbs to 40,000 ft altitude and accelerates to 1000 ft/sec velocity. This velocity is sufficient to ensure enough thrust from the ramjet engines to overcome the drag,

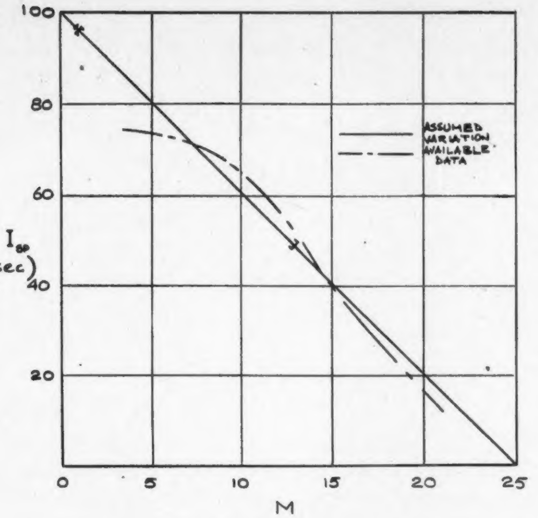

Figure 3
I_{sp} vs Mach number

and the altitude is high enough to prevent sonic boom damage on the ground. At this point the ramjets are lit, and separated from the first stage. From this altitude and speed, the first stage may easily be recovered by parachute or glide descent.

The ramjet powered second stage is an arrowhead-shaped hypersonic aircraft. It is recoverable and is propelled by hydrogen burning ramjets. Its payload consists of the third stage rocket. The I_{sp} (Figure 3) and sfc (Figure 4) are based on the following assumptions:

(1) absolute entropy increase in intake $\quad S/R_0 = 0.7436$

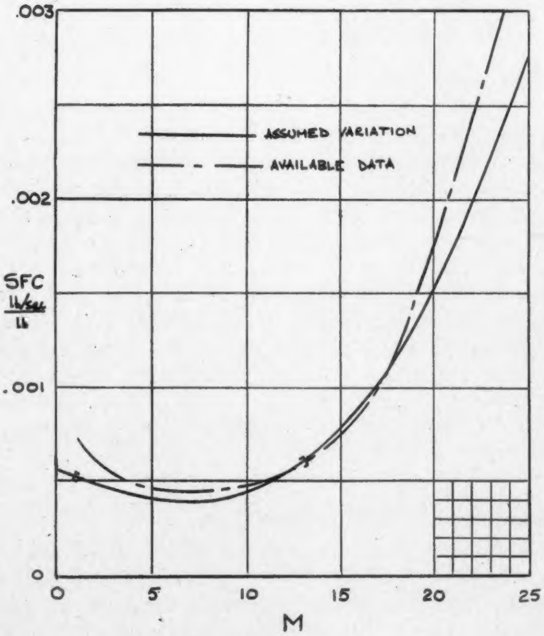

Figure 4
sfc vs Mach number

(2) nozzle velocity coefficient $\quad$ 0.96
(3) fuel $\quad$ stoichiometric hydrogen
(4) combustion efficiency $\quad$ 100%

The task of the ramjet is to achieve a maximum increase in velocity. It is desired that it accelerates from 1000 ft/sec to 13,000 ft/sec. While doing this the ramjet is required to climb from 40,000 ft to 90,000 ft altitude (Point 2 in Figure 2). This climb entails a loss in thrust because of the density decrease with altitude; it is necessary, however, to avoid the "Too Hot" region (Figure 2).

The third stage is a ballistic rocket; it is expected to take its payload from a velocity of 13,000 ft/sec and 90,000 ft altitude to orbital velocity at 1,100,000 ft. A Vanguard type satellite in a circular orbit at this altitude has a life span of 75 days[8].

First stage

Simple performance calculations showed that the first stage would be capable of accelerating from take-off to 1000 ft/sec while climbing to 40,000 ft. During this acceleration climb the first stage would consume 15% of its own weight as fuel. This is within the present day capabilities of both turbojet and rocket powerplants.

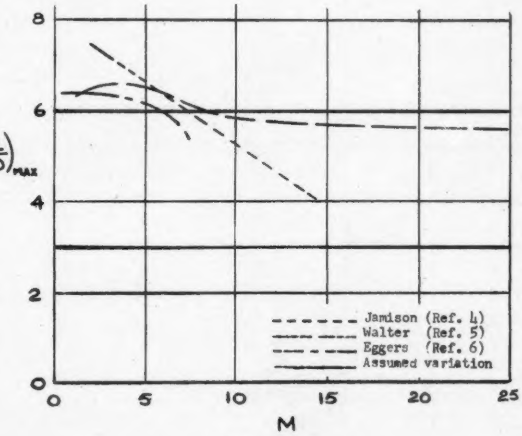

Figure 5
Forces acting on a powered lifting vehicle flying in an atmosphere

Second stage

Consider Figure 5, which shows a thrusting, lifting, and dragging vehicle in motion in a central force field. Resolving forces parallel and perpendicular to the direction of motion yields:

$$\frac{W}{g}\frac{dV}{dt} = \mathscr{T}\cos\alpha - D - W\sin\phi \quad (1)$$

$$\frac{W}{g}V\left[\frac{d\phi}{dt} - \frac{V}{r}\right] = L - W\cos\phi + \mathscr{T}\sin\alpha \quad (2)$$

These equations and their various simplifications form the basis of most trajectory calculations[7].

The second stage trajectory is not easy to calculate, because it involves a pronounced decrease in weight and a thrust which varies with the vehicle speed and ambient air density ρ. In addition, the sfc and I_{sp} are functions of Mach number (see Figures 3 and 4). For

these reasons the ramjet trajectory calculation was performed on an IBM 650 digital computer. The simplifying assumptions in this calculation were:

(1) the L/D ratio remains constant throughout the flight at either 6.0 or 3.0,
(2) the climb angle ϕ is small so that the equations of motion may be linearized,
(3) the angle of attack α is negligibly small,
(4) the earth is spherical and non-rotating,
(5) altitude of vehicle is small compared with the radius of the earth, and
(6) the intake or capture area of the ramjet A is constant.

Combining Eqs. (1) and (2) by making use of assumptions (1) and (2):

$$\frac{W}{g} V\left[\frac{d\phi}{dt} - \frac{V}{r}\right] = C\left[\mathscr{T} - \frac{W}{g}\frac{dV}{dt} - W\sin\phi\right] - W\cos\phi \quad (3)$$

Dividing by W,

$$\frac{V}{g}\left[\frac{d\phi}{dt} - \frac{V}{r}\right] = C\left[\frac{\mathscr{T}}{W} - \frac{1}{g}\frac{dV}{dt} - \sin\phi\right] - \cos\phi \quad (4)$$

The following geometrical relationship exists between h, V and ϕ:

$$\frac{\frac{dh}{dt}}{V} = \sin\phi \quad (5)$$

this allows ϕ in Eq. (4) to be converted to altitude and velocity. Eq. (4) is true for any vehicle moving along any h-V path. If we now specify the path in terms of h and V as,

$$h = h(V), \quad (6)$$

then

$$\frac{dh}{dt} = \frac{dh}{dV}\frac{dV}{dt} = h'\dot{V} \text{ where } h' = \frac{dh}{dV} \quad (7)$$

For small ϕ (assumption 2)

$$\phi \approx \sin\phi = \frac{h'\dot{V}}{V} \quad (8)$$

and

$$\cos\phi \approx 1. \quad (9)$$

From Eq. (8) we get, by differentiation,

$$\frac{d\phi}{dt} = \dot{\phi} = \frac{h'}{V^2}(V\ddot{V} - \dot{V}^2) \quad (10)$$

Substitution of Eqs. (8), (9) and (10) into Eq. (4) yields the linear equation of motion along a given path in the h-V plane defined by $h = h(V)$,

$$h'\ddot{V} - \frac{h}{V}\dot{V}^2 + C\left[1 + \frac{gh'}{V}\right]\dot{V} - \frac{V^2}{r} - Cg\frac{\mathscr{T}}{W} + g = 0 \quad (11)$$

This is a second order differential equation in V and can be solved as such by numerical methods provided that, over the integration interval, the variations of C, g, r and $1/W$ are small. The equation becomes more complicated if the above four quantities are allowed to vary with V.

It is of interest to note that in the equation of motion (Eq. (11)) $\mathscr{T}$ and W appear only once and then as a ratio. This thrust/weight ratio is equal to

$$\frac{\mathscr{T}}{W} = \frac{\rho V I_{sp} A}{W_i - \int_0^t \mathscr{T}.\text{sfc.}\, dt}$$

$$= \frac{\rho V I_{sp} A}{W_i - A \times \int_0^t \rho V.I_{sp}.\text{sfc.}\, dt}$$

$$= \frac{\rho V I_{sp}}{\frac{W_i}{A} - \int_0^t \rho V I_{sp}.\text{sfc.}\, dt}$$

If the intake area A remains constant, then, for a given vehicle, W_i/A is a constant. Since neither W_i nor A appear anywhere else in the equation of motion it must be concluded that vehicles with equal W_i/A have the same performance. Thus W_i/A may be used as a performance parameter or scaling factor for ramjet powered vehicles.

For the problem at hand, a straight line path in the flight corridor was arbitrarily chosen. Since no trajectory optimization of any kind has been attempted, there is no reason to suppose that this straight line path is in any way the best path.

The I_{sp} and the sfc for the engine characteristics, outlined previously, are shown in Figures 3 and 4, the dashed lines indicate available data and the solid lines indicate values used in the present computations. Brackets on the solid lines show the range of values within which computations were performed. Reference 8, from which this data was obtained, shows that I_{sp} and sfc are very weak functions of altitude. For this report, therefore, they were assumed to be independent of altitude.

Third stage

Using hydrogen-oxygen for fuel, the specific impulse of the third stage was taken as 400 sec. It started at 90,000 ft with a velocity of 13,000 ft/sec (final ramjet conditions) and climbed to 1,100,000 ft while accelerating to 25,500 ft/sec. The orbital speed at this altitude is 25,300 ft/sec. It was noted that the velocity and radius vectors were not quite perpendicular. This would result in an elliptical orbit. The life span of a Vanguard type satellite at this altitude is 75 days. Burning time of the rocket was 100 seconds and the thrust vector inclination ψ to initial horizontal varied according to $\psi = \tan^{-1}(6.0 - 0.03t)$. The equations for the third stage were integrated using a method described in Reference 9.

RESULTS AND DISCUSSIONS

Figure 2 shows the flight of the proposed system in the flight corridor. The flight commences with a rocket powered ground roll from the Point 0. The ground roll is followed by an accelerating climb to 1. It is assumed that the rocket stage, which is required to accomplish the transition from 0 to 1, has a struc-

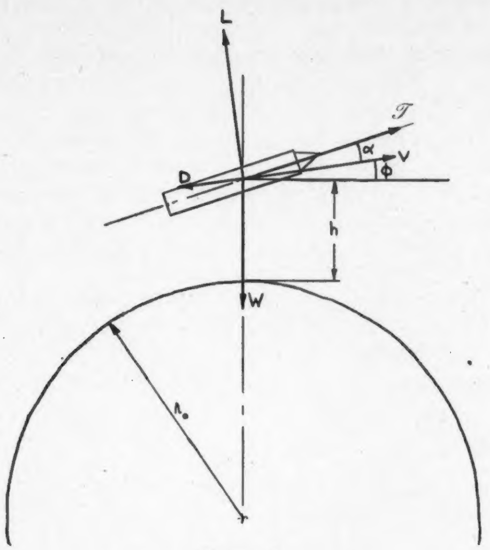

Figure 6
***L/D* ratio vs Mach number**

tural weight which is 15% and a fuel weight which is 15% of the initial all-up weight.

At Point 1, the empty rocket cluster is released and the ramjets are ignited. The ramjet vehicle then proceeds to Point 2. During its flight the I_{sp} varies as shown by the solid line in Figure 3. This line is given by, $I_{sp} = 100 - 4M$. This line is higher than the available data (dashed line) in the velocity range 1000 to 7200 ft/sec and lower in the range 7200 to 14,500 ft/sec.

The sfc is shown in Figure 4. The dashed line corresponds to available data and the solid line is a polynomial of the third order which approximates the available data. The solid line is given by

$$\text{sfc} = 5.556 \times 10^{-4} - 4.35 \times 10^{-5} M + 1.620 \times 10^{-6} M^2 + 1.50 \times 10^{-7} M^3$$

Over most of the velocity range of interest (1000 to 13,000 ft/sec) the polynomial gives lower than actual values for the sfc.

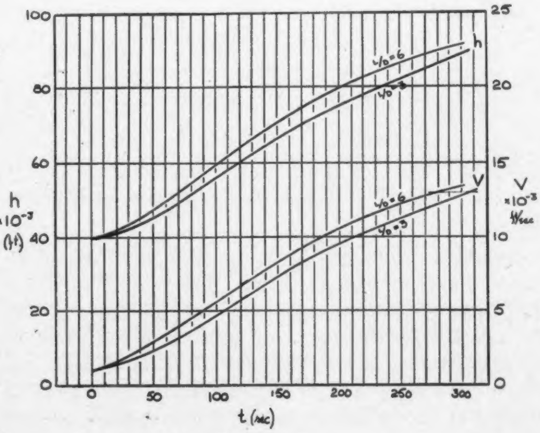

Figure 7
Altitude and velocity vs time

Figure 6 shows the variation of maximum L/D ratio with Mach number as obtained from various references. Underwing propulsion combustion[10] promises even higher L/D ratios. For the present report two trajectories were calculated, one having an L/D ratio of 6.0 and the other having 3.0. The L/D ratios were assumed to be independent of Mach number. Judging from the reference curves, an L/D ratio of 6.0 would represent sophisticated aerodynamic design, whereas 3.0 would be indicative of mediocre to poor design. This is especially true for the lower Mach number range.

Figures 7 to 11 show the results of the trajectory calculations for the ramjet powered second stage. Figure 7 is a plot of altitude h and velocity V against time from separation of the ramjet fom the rocket cluster t for L/D ratios of 6.0 and 3.0. It is worth noting that the altitude and velocity graphs are similar. This is due to the specification of the linear path in the flight corridor given by, $h = 35{,}833 + 4.1667\ V$. This path joins the points 40,000 ft, 1000 ft/sec and 90,000 ft, 13,000 ft/sec, respectively. Further it is seen that a vehicle with $L/D = 3.0$ reaches the terminal

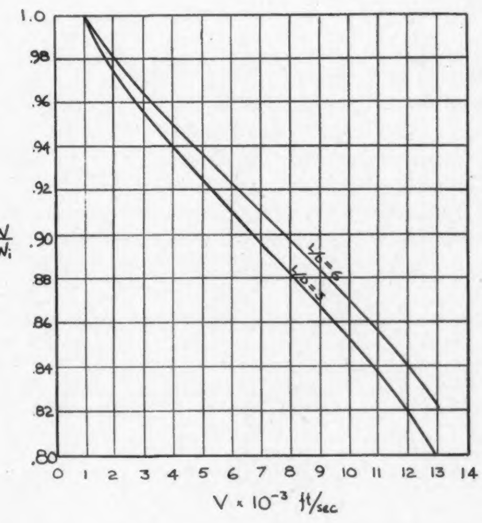

Figure 8
Weight fraction vs velocity

condition (Point 2) 35 seconds later than one with $L/D = 6.0$. The slope of the height curve is the rate of climb; it starts at $t = 0$ sec with a value of 98.41 and 71.13 ft/sec, reaches a maximum of 248.39 and 225.10 ft/sec at 90 and 105 sec and then drops to 109.30 and 93.69 ft/sec at $t = 280$ and 320 sec, for an L/D of 6.0 and 3.0, respectively. The slope of the velocity curve ($dV/dt = a$) is plotted in Figure 10.

Figure 8 shows the weight fraction of the second stage (W/W_1) versus velocity, for $L/D = 6.0$ and 3.0. The curve starts at 1000 ft/sec with $W/W_1 = 1.0$, it then drops to 0.8225 and 0.7975 at 13,000 ft/sec. This means that in order for a ramjet powered vehicle to accelerate from 1000 ft/sec to 13,000 ft/sec at $L/D = 6.0$ and 3.0, respectively, it must have a fuel supply equivalent to 17.75% and 20.25% of its initial

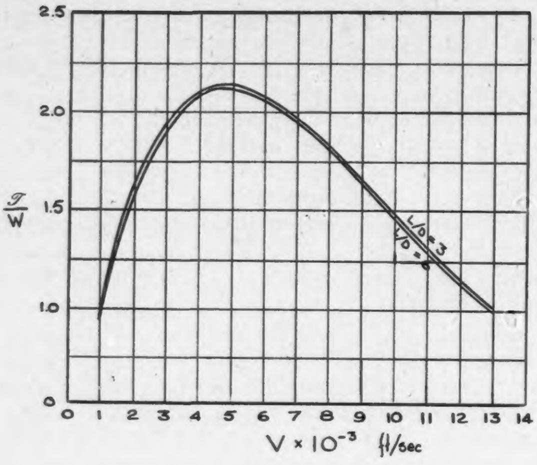

Figure 9
Thrust/weight ratio vs velocity

all-up weight. It becomes apparent also that the $L/D = 3.0$ vehicle consumes,

$$\frac{20.25 - 17.75}{20.25} \times 100 = 12.5\%$$

more fuel than the $L/D = 6.0$ vehicle.

Figure 9 is a plot of the $\mathcal{T}/W$ ratio against velocity. The $\mathcal{T}/W$ ratio starts at a value near 1.0 at 1000 ft/sec and reaches a maximum of 2.124 and 2.147 at 4730 and 4731 ft/sec. It then again drops to a value near 1.0 at 13,000 ft/sec. It is noted that the $\mathcal{T}/W$ curve for $L/D = 3.0$ is higher than that for $L/D = 6.0$. This is apparent considering that thrust depends only on velocity (not on L/D ratio) and that weight decreases more rapidly with velocity at the lower L/D ratio.

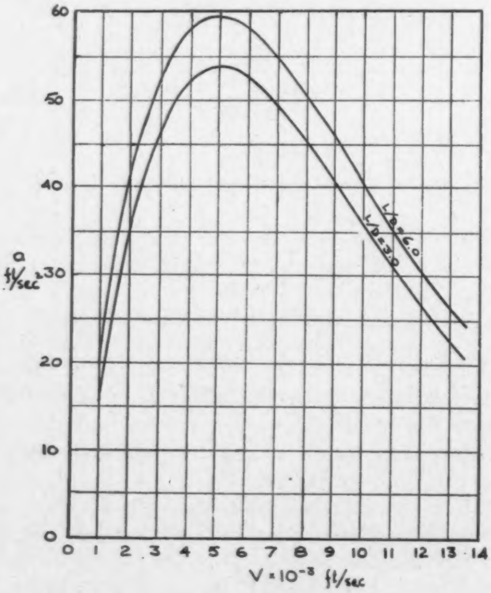

Figure 10
Acceleration along the flight path vs velocity

Figure 10 shows the variation of the acceleration in the direction of the flight path with velocity. The acceleration reaches a maximum of 59.61 and 54.02 ft/sec² at 5029 and 5002 ft/sec. This acceleration (less than 2g) is well within human tolerances.

In addition to time t, velocity V, acceleration a, altitude h, $\mathcal{T}/W$ ratio, and weight W, the computer calculated also the following variables at each point of the trajectory:

rate of climb	dh/dt
climb angle	$\phi = 1/V \; dh/dt$
rate of change of climb angle	$d\phi/dt$
Mach number	M
fuel flow	dw/dt
specific impulse	I_{sp}
thrust	$\mathcal{T}$

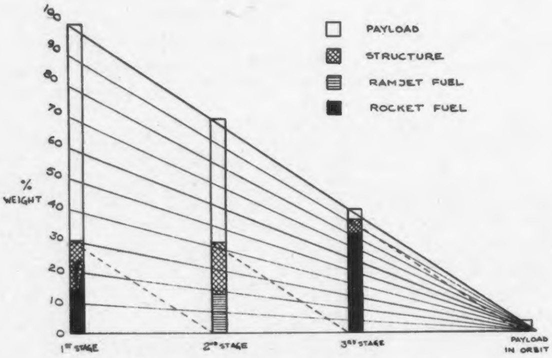

Figure 11
Stage weight breakdown

The following table gives the starting value, the maximum and the final value of the rate of climb, the climb angle and the rate of change of climb angle for the ramjet trajectory ($L/D = 6.0$):

	Start	*Maximum*	*End*
$\frac{dh}{dt}$ ft/sec	98.41	248.39	109.30
$\frac{dh}{dt}$ ft/min	5900	14,900	6,560
ϕ deg	5.129	5.147	0.4803
$\frac{d\phi}{dt}$ deg/sec	0.008812	0.008812	−0.002980

The small values of ϕ provide a justification for the linearization of the equations of motion made previously.

Figure 11 summarizes the capabilities of a rocket-ramjet-rocket boosting system. The rocket powered first stage consists of 15% fuel, 15% structure and 70% payload. This payload makes up the ramjet powered second stage of which 18% is fuel, 25% is structural weight and 57% is payload. This payload constitutes the rocket propelled third stage of which 80% is fuel, 12% is structure and 8% is payload. This is the payload that the system is capable of orbiting.

The over-all payload percentage (i.e. payload in orbit/initial gross weight × 100) is 3.25%. It is apparent from Figure 11 that, compared with the first and second stages, the third stage carries very little payload. Thus, attempts at achieving improvements should be directed first towards the third stage. It is believed that a better payload in orbit could be achieved if the second stage would be allowed to reach a speed of the order of 20,000 ft/sec at 200,000 ft altitude. This is within the capabilities of the type of ramjet considered in this report. The rocket would then increase altitude by 900,000 ft and velocity by 5,000 ft/sec. An estimated payload in orbit equal to approximately 8% of the initial gross weight may be achieved by this method.

CONCLUSION

A rocket-ramjet-rocket type of composite launcher is suitable for boosting into orbit 3.25% of its initial gross weight. It is estimated that stage and flight path optimization may raise this figure to 8%.

Extreme sophistication in aerodynamic design of this accelerating vehicle is unnecessary, since a vehicle with an L/D ratio of 3.0 gives only 0.2% less payload in orbit than one with an L/D ratio of 6.0.

The first stage, consisting of a cluster of rockets, may be recovered by parachute and re-used. The second stage may be returned for landing and then re-used.

The accelerations experienced during the whole boost phase are of the order of 2g and thus well within human tolerances.

In comparison with pure rocket boosters the ramjet system is lighter, uses less fuel and is completely recoverable. For equal initial gross weights the ramjet puts into orbit a payload which is 30 times larger than that achievable with rocket launchers.

The analysis shows that two ramjets having equal values of W_1/A have the same performance.

It is to be noted that the results obtained are valid for any ramjet vehicle whose ratio of initial weight to intake area (or capture area) is 1000 lb/ft^2; this implies that we may use a 1000 lb vehicle with a 1 sq ft capture area or a 300,000 lb vehicle with a 300 sq ft intake area, to give the same performance. The smaller vehicle is probably the lower limit on weight because of mechanical size restrictions and the larger is the upper limit because of prohibitively large capture area requirements. A practical vehicle has a weight somewhere between these limits.

ACKNOWLEDGEMENT

The instigation by, and subsequent advice from, Dean D. L. Mordell is gratefully acknowledged.

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APPENDIX

NUMERICAL SOLUTION OF THE RAMJET VEHICLE'S EQUATION OF MOTION

The second order differential equation in V (Eq. (11)) was converted into two first order simultaneous equations in a and V by the substitution

$$a = \dot{V} \tag{14}$$

Eq. (11) then becomes

$$\dot{a} = \frac{a^2}{V} - C\left[\frac{1}{h'} + \frac{g}{V}\right]a + \frac{V^2}{h'r} + \frac{Cg\rho VAI_{sp}}{h'W} - \frac{g}{h'} \tag{15}$$

Eqs. (14) and (15) are then the two first order simultaneous equations. In the numerical integration of these equations $C = 3.0$ or 6.0, $g = 31.91$ ft/sec^2, $r = 21.21 \times 10^6$ ft, $A = 1.0$ ft^2.

Since the effect of variation of W is an integrated one, it was thought that sufficient accuracy would be achieved by using the average weight between two points of integration, i.e., the weight between any two points 1 and 2 was taken as,

$$W = W_1 - \frac{\text{sfc}.\mathscr{T}.\Delta t}{2}$$

The Runga-Kutta Method was used to solve Eqs. (14) and (15). The solution was programmed in the McGill Fortran Automatic Coding System and performed on an IBM 650 digital computer at the Computation Centre, McGill University.

The program computes any monotonically climbing or descending trajectory in the h-V plane that can be specified by a third order polynomial of the form.

$$h = c_0 + c_1V + c_2V^2 + c_3V^3.$$

Similarly, the I_{sp} and the sfc may be specified by,

$$I_{sp} = e_0 + e_1M + e_2M^2 + e_3M^3$$
$$\text{sfc} = b_0 + b_1M + b_2M^2 + b_3M^3.$$

CANADIAN AERONAUTICAL INSTITUTE

SOME ASPECTS OF THE SPACE COMMUNICATION PROBLEM†

by R. Smelt*

Lockheed Aircraft Corporation

SUMMARY

Communication from satellites and interplanetary space vehicles is a logical extension, to longer ranges, of present day radio communication systems on earth. It is not practicable to achieve the required increase in range by increasing transmitter power in the space vehicle; but there are developments offering reduction of receiver noise and increases in antenna gain, both on the ground and in the satellite. Progress is being made on construction of very large ground antennas, and on development of unfurlable antennas for the space vehicle. Both appear to be limited in their application by the requirements for accurate alignment to the space vehicle position and orientation.

Present techniques employed for communication with low altitude satellites in the Discoverer program are illustrated, and the effect of improvements on communications for typical satellite and interplanetary missions is presented in a table. With all the improvements which currently appear feasible, the theoretical maximum information rate for a Mars or Venus probe is only about 2500 bits/sec, and in practice only about one quarter of this maximum may be realized. The paper draws attention to the possibility of relay stations in interplanetary space, utilizing stable equilibrium positions in the moon-Earth and Earth-sun gravitational fields similar to the Trojan satellites.

INTRODUCTION

RADIO communication was a well developed technology at the time the first artificial satellites were placed into orbit. The provision of communication channels to and from the satellites for data retrieval and, later, for command, has been largely a matter of adapting these techniques, and fitting them to the special requirements and environments of space. Satellite development is now moving into a second phase, in which applications to meteorology, surveillance, communication relaying or navigation on earth are each posing their special requirements on the radio communication links. In addition, lunar and interplanetary space vehicles now require communication systems operating over ranges of millions of miles. Nevertheless, all these developments still represent logical extensions of the present radio communication systems on earth, and it is instructive to make a direct comparison between the new space systems and the established terrestrial radio links.

A representative earth-bound system is the TH1 microwave radio relay, which now carries an appreciable fraction of the wide-band communication traffic in the USA and Canada. A typical relay in this chain operates with a transmitting power of about 5 watts, of the same order of magnitude as typical present day satellite transmitters; the TH1 frequency, around 6 kmc/sec, is also fairly close to the optimum for space-to-earth transmission, as will be seen below. The most obvious difference is in the range of operation. The ground relay stations are usually located on hilltops about 30 miles apart, but even the lowest altitude earth satellites (e.g., the Discoverer series, with apogees of about 100 nautical miles) require communication over a range of about 1,000 miles to the horizon. The "stationary" satellites, moving on the equator with the same rotational period as the earth, at a height of 22,000 miles, are of considerable interest for application as intercontinental communication relays; they require transmission paths exceeding 25,000 miles. Deeper space probes are required to transmit information over interplanetary distances measured in millions of miles. Since the power received is inversely proportional to the square of the distance in free space, it is evident that the space vehicle communication system must be capable of operating satisfactorily with received signal strengths many orders of magnitude lower than corresponding receivers of the ground-to-ground system.

One obvious method of mitigating this problem is to increase the transmitted power. In the frequency range of interest for space communication applications, output tubes are now available which can supply RF power measured in kilowatts. These tubes can be used in ground stations as a source of transmitted power to the satellite or space vehicle. Their use in the inverse direction, from satellite to ground, presents a number of formidable problems.

In the first place, such high power tubes are heavy and bulky but, even more important, the provision of power on the satellite to feed the RF generator is exceedingly difficult and expensive. If operation for more than a few days is desired, solar power is the only feasible primary source which has been used up to the present. In the vicinity of the earth, the sun provides ample power for communication purposes, but the collector required to harness this power with

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†Paper read at the Joint I.A.S./C.A.I. Meeting in Montreal on the 17th October, 1960.

*Chief Scientist, Missiles and Space Division

Figure 1
60 ft antenna of Discoverer Station, Hawaii

present techniques costs $500/watt, and weighs about ¼ lb/watt. Since it currently costs about $1,000 to place 1 lb of payload into even a low altitude orbit, it will be evident that the provision of large amounts of power by solar collectors is inordinately expensive at present. It is possible that future improvements may significantly reduce the cost, or that the development of nuclear auxiliary power sources may provide greater economy. Even if these developments take place, however, it is doubtful whether high power transmitters would then be used in space vehicles, because of the reliability problem.

In practically every satellite or space vehicle application, the active components within the vehicle are required to continue to function satisfactorily for long periods without maintenance. The current target for most satellite payloads is about one year of operation. The interplanetary vehicles now in the planning stage will require electronic equipment with similar life performance, since the typical interplanetary journey will take many months or even years to execute. On a more utilitarian note, the economics of a satellite communication relay depends primarily upon vehicle lifetime, and equipment with a useful operating life of ten years is under study and development. It is currently considered that long life and high power are not compatible in microwave tubes, since high powers imply high emission currents which are synonymous with short life in conventional vacuum tubes. It is conceivable that the development of high power solid state devices operating in the microwave region may completely change this picture but, for the time being, it appears that, from considerations of reliability as well as primary power supply, satellite transmitter output will be limited to a maximum of a few watts.

The current problem of space communication is thus primarily in the vehicle-to-ground link, and can be stated very simply. With roughly the same power as now used for distances of about 30 miles on the ground, how is it possible to cover satellite distances of thousands of miles, or interplanetary distances of millions of miles?

ANTENNAS

The first step in achieving the best possible conditions for space communication is to utilize the maximum practicable gain in both the transmitting antenna on the space vehicle and the receiving antenna on the ground. The ratio of received power P_R to transmitted power P_T is given by the familiar range equation

$$\frac{P_R}{P_T} = \frac{G_T\, G_R\, \lambda^2}{(4\pi)^2\, R^2} \qquad (1)$$

where R is the range in free space
λ is the wavelength
and G_T, G_R are antenna gains defined by

$$G = \frac{4\,\pi\, A}{\lambda^2}$$

where A is the effective or "capture" area of the antenna, slightly less than its physical area.

There are engineering limitations to the gain which can be employed in both the satellite and ground antennas; determined in both cases by the space available and the problem of accurate alignment. Considering first the simpler case of the ground antenna, movable dishes have always been used up to the present, although electronically steerable ground arrays of greater area appear feasible and may, in fact, be used for communication links with pseudo-stationary satellites on 22,000 mile orbits, where the movement of the beam is restricted to a small sector of the sky. In the low altitude Discoverer program, dishes 60 ft in diameter have been employed and have usually been supplemented by tri-helix antennas; the installation at Kaena Point in Hawaii is shown in Figure 1. Installations using a combination of these types of antennas, with suitable tracking equipment, have been placed at

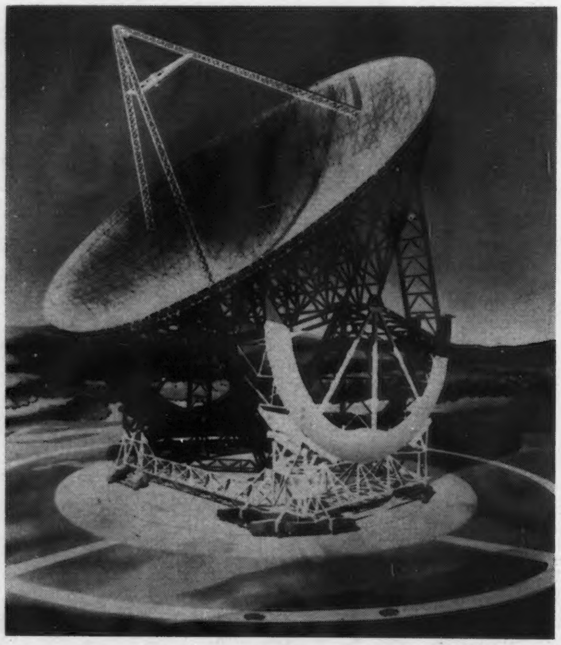


Figure 2
Artist's view of the 600 ft antenna being constructed at the US Naval Research Station

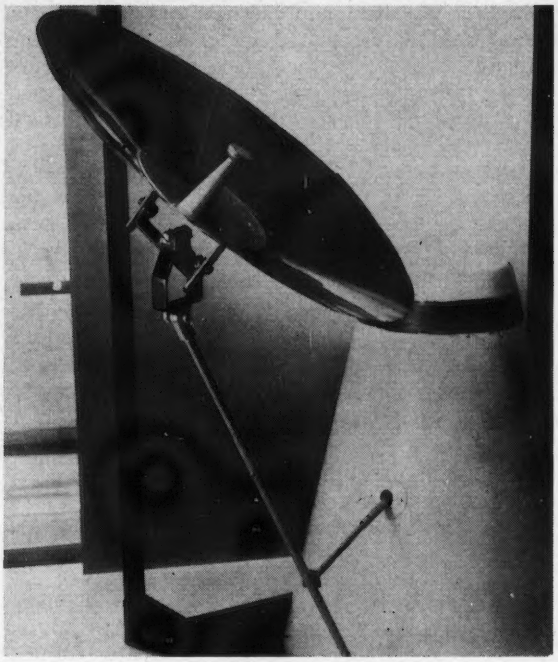


Figure 3
Vane antenna — open

a number of points around the globe selected for best interception of polar orbits.

Although the 60 ft dish has proved very satisfactory for Discoverer satellites, the deep space probes and lunar vehicles launched up to the present have depended largely upon reception by the 250 ft diameter dish at Jodrell Bank in England. This is the largest receiving dish in the world at present; however, it will shortly be dwarfed by the radio telescope now being built at the US Naval Radio Research Station in Sugar Grove, West Virginia. This dish is 600 ft in diameter and more than 7 acres in area. Figure 2 is a photograph of a model of the dish, furnished by North American Aviation, Inc.

The major problem in dishes of such large diameter is that the beam is extremely narrow. Approximately, the beam angle θ measured between the half power points in the radiation path is given by $\theta = \lambda/D$ where D is the dish diameter.

For a transmission frequency of 6 kmc/sec this implies a beam angle of 0.2° with a 60 ft antenna, and only 0.02° with the 600 ft Sugar Grove antenna. It will be evident that as the diameter of the receiving antenna increases, the structure becomes more unwieldy to move and, on the other hand, the requirements for accurate alignment to the space vehicle position become more difficult. In fact, the basic component of the direction system of the Sugar Grove antenna is an inertial guidance system utilizing ballistic missile guidance techniques. There is, of course, a difficult engineering problem in providing the necessary precision in the dish shape, and in ensuring that the deflections of the dish under its weight and wind forces do not destroy the gain.

The satellites and space vehicles launched up to the present have used comparatively simple antennas, primarily because of the problems of directing a high gain, narrow beam system on a satellite. Most space vehicles flown up to the present have had no stabilization devices, so that it was imperative to use a substantially isotropic transmitting antenna. Some satellites have been spun in orbit, but this simple stabilization system does not maintain its orientation relative to earth stations, so that again its antenna must be almost isotropic. The Agena satellite, as used in the Discoverer series, is stabilized with an accuracy of about 1° relative to local earth axes, but again high directive antennas are not used because of the desire to receive signals from it at a number of stations with different orientations (combined with an understandable desire, in the early flights, to receive information from the vehicle in the event that the stabilization system did not work adequately!). For most applications of the Agena satellite now contemplated, the communication link is adequate without high gain in the satellite's transmitting antenna. We are, however, in the research organization of LMSD, investigating methods[a] of adding high gain antennas which can be directed to a ground receiving station, taking advantage of the stabilization feature of the Agena vehicle. These methods depend upon the fact that the forces on the antenna in orbit, from its weight or its air loads, are completely negligible, so that the antenna need only be strong enough to carry the small forces required to direct it. We have investigated mechanical vane devices, shown in the open position in Figure 3. The alternative possibility of unfurling

Figure 4
Aluminum foil antenna — open

[a]Supported in part by the Wright Air Development Division of the Air Research and Development Command, under Contract No. AF 33(616)-6022.

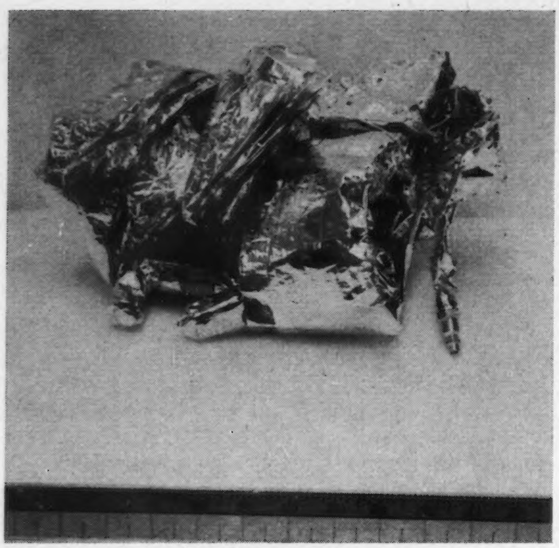


Figure 5
Aluminum foil antenna — packed

their structure by air pressure may ultimately be still more attractive. The Echo passive satellite is the first practical application of this technique to a satellite in orbit, with the simplest possible inflatable structure, a sphere. A more sophisticated antenna for an active transmitter with appreciable gain is shown in Figures 4 and 5 in the open and folded positions. It is important to emphasize here that the opening process is not merely a matter of inflation, but rather a matter of placing the aluminum foil elements in the correct direction, and smoothing out the kinks, so that they can carry the loads required to direct them. The inflating gas is, in fact, expelled after the antenna is erected.

These antennas are examples of a number of alternative configurations which have been investigated at LMSD. A more complete account of our work in unfurlable antennas has been given by P. D. Kennedy in a conference at the University of Illinois. It appears to us that unfurling techniques will permit the transmitting antenna in a satellite to have a gain commensurate with accuracy of stabilization and direction which can be incorporated in the satellite's control system; the ultimate limit is then set, exactly as in the ground antenna, by the precision of the guidance system.

At this point, we can again compare the space communication system with the ground-to-ground microwave link, which usually employs dishes about 6 ft in diameter. The 60 ft dishes of present satellite nets represent a gain of 100, and the US Navy antenna shown in Figure 2 will provide a healthy improvement of 10^4 compared with the receiving antenna of the ground system. On the other hand, the almost isotropic transmitting systems of our present satellites and space probes imply a serious loss in efficiency compared with the 6 ft ground transmitting antenna, which has a gain of about 10^4. In other words, even if we employ the 600 ft dish shown in Figure 2, with a non-directional satellite antenna, the over-all performance is no better than the present antenna system in ground-to-ground links, and we must look for other improvements to handle the longer ranges in space. If, on the other hand, we can unfurl and direct a 6 ft antenna on the space vehicle (its beam width of about 1½° at 6 kmc/sec would be consistent with stabilization accuracies already attained on the Agena vehicle), then we can increase the gain by a factor of 10^4, or the range by a factor of 100 for the same communication bandwidth and quality.

NOISE

For the extended ranges of satellite and interplanetary communication, it is necessary to accept lower received power levels than in present terrestrial communication links, and this implies that the masking noise in the receiver must be reduced. As is well known, this noise is partially generated in the receiver itself, and partially contributed by outside sources, in the atmosphere or further distant in space. Both these components have had considerable investigation, and are discussed in recent literature (e.g., J. R. Pierce and C. C. Cutler have an excellent discussion of interplanetary communications in Vol. I of *Advances in Space Science*). It is therefore sufficient to give a brief summary here. As is conventional, noise power components will be expressed in terms of an equivalent noise temperature T_N°K; for a pre-detection bandwidth of B cycles/sec, the noise power is kT_NB watts, where k is Boltzmann's constant, 1.380×10^{-23}.

Figure 6, which is taken, with minor modifications, from the article by Pierce and Cutler, shows how the average galactic noise increases with decreasing frequency. On the other hand, the noise due to atmospheric absorption and reradiation (mainly from water vapor and oxygen) becomes significant at frequencies above about 10 kmc/sec. It follows, as Figure 6 clearly demonstrates, that a space-to-ground link has minimum external noise in the frequency range from about 3 to 10 kmc/sec. In recent Federal Communications Commission hearings in Washington (August 1960) a number of organizations interested in space developments have stressed the desirability of special fre-

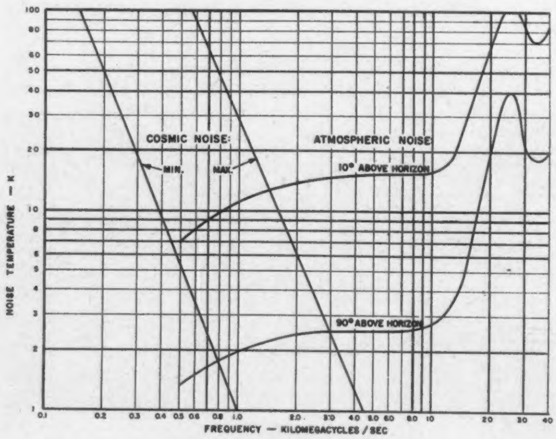


Figure 6
Cosmic and atmospheric noise

quency allocations in this part of the spectrum by space communications. It is important to stress that Figure 6 does not include man-made noise, or peaks of galactic noise which would occur, for example, if the antenna were pointed directly at the sun. In ground-to-ground communication links such as the TH1 net, a very generous margin is included for man-made interference, but this margin cannot be afforded in space communications. Fortunately, this large margin is not so necessary, since it is possible to locate the receiving antenna in relatively isolated areas and to take precautions to minimize interference pickup by the antenna side lobes.

A further point which is noteworthy in Figure 6 is that for vehicle-to-vehicle communications above the atmosphere, the upper frequency limit at about 10 kmc/sec does not apply. In fact, it appears desirable to move into those RF regions which are almost completely absorbed by the atmosphere, so as to have a natural shield against interfering signals from the earth. A number of workers in the field have therefore suggested that the infra-red, X-ray and gamma-ray regions of the spectrum might be advantageous for communication entirely in space. Unfortunately, the efficiency of signal generation and the sensitivity of detectors in these regions are far from attractive, currently, in comparison with the more conventional RF bands. As an example, with present day techniques in infra-red and optical detection, transmission over lunar distances would require extremely high signal powers (modulated light beams of the order of 10 kw) and apertures of about 40 inches, demanding extreme precision in alignment. Furthermore, our current knowledge of the amount of radiation in space in these frequency bands is not yet adequate; IGY data obtained in rocket experiments by the US Naval Research Lab, for example, show very high radiation intensity near the Lyman-α band.

The last few years have seen a considerable improvement in the noise generated within the receiver itself. Table 1, again provided by Pierce and Cutler, shows the large reduction in noise temperature which has been obtained by parametric amplifiers and masers. The 85 ft antenna of the National Aeronautics and Space Administration at Goldstone in California employs a parametric amplifier, and its noise temperature is reported to be about 100°K. The special horn antenna, which Bell Telephone Labs has developed for the Echo program, employs a maser detector, and the total noise temperature is around 20° to 25°K. These developments imply a gain exceeding 100 in signal-to-noise ratio, or more than a ten-fold increase in range for the same signal bandwidth and quality.

INFORMATION CAPACITY

We can now examine how the developments in antennas and detectors, summarized above, will improve the capacity of space communication systems, i.e., will permit a wider bandwidth to be handled with a given signal-to-noise ratio. Specifically, the range equation (Eq. (1)) leads directly to a value for the product of bandwidth B and signal-to-noise ratio S/N:

TABLE 1

Type of Receiver	Noise Temperature (°K)
Crystal mixer, $\lambda < 1$ cm	10,000
Crystal mixer, $\lambda > 5$ cm	1,500
Traveling-wave tube, $\lambda > 5$ cm	750
Parametric amplifier, $\lambda > 5$ cm	100
Maser, $\lambda > 5$ cm	10
Maser, expected future improvement	3

$$B \,.\, S/N = \left(\frac{\lambda}{4\pi R}\right)^2 G_T \, G_R \, \frac{P_T}{kT} \tag{2}$$

where T is the total receiver noise temperature.

The quantity $B\,.\,S/N$ is a figure-of-merit for the communication link; in fact, Pierce has pointed out that for space-to-ground links, where small vehicle power forces the utilization of large bandwidths for even moderate information capacity, the quantity $B\,.\,S/N$ measures directly the theoretical maximum rate I_{MAX} at which information can be sent over the link:

$$I_{MAX} = 1.44\, B \,.\, S/N \text{ bits/sec.} \tag{3}$$

Table 2 gives values of this basic quantity I_{MAX} for a number of existing or projected space communication links. For simplicity, in preparing this table, it has been assumed that all links are operated at the same frequency of 6 kmc/sec. The earlier satellite-to-ground links have usually been operated at somewhat lower frequencies, in the VHF bands, so that their performance was somewhat inferior to the numbers given in the table. It is evident, however, that the low altitude satellites do not pose any serious communication problem; even with present day equipment, the maximum theoretical information rate is almost 10^{10} bits/sec, which more than satisfies the data transmission requirements of such satellites, even with very inefficient use of the system from the information coding point of view.

It must be emphasized that the theoretical information rate given in Table 2 represents an ideal upper limit which is not attained in practice. A discussion of communication theory and the superiority of certain modulation and demodulation techniques is beyond the scope of the present paper, and it is sufficient to give a brief summary. The familiar amplitude modulation of a carrier determines the bandwidth directly in terms of the information content (speech, TV etc) and is usually far from the optimum bandwidth for minimum power. In consequence, to obtain voice or TV transmission of adequate quality, the information rate must be 10^2 to 10^3 lower than the theoretical maximum. Some improvement is possible by suppressed carrier or SSB techniques; but FM transmission is much to be preferred for such applications, since its information content can approach within a factor of 20 the theoretical value. Still better utilization can be obtained by adopting feedback systems such as phase lock or FB/FM, which can approach within a factor of 3 or 4 the full theoretical performance. Needless to say, such phase lock systems have

TABLE 2

TYPICAL PERFORMANCE OF SPACE-TO-EARTH COMMUNICATION LINKS

(ASSUMING COMMON RF OF 6 KMC/SEC)

Application	Maximum Range	Transmitter		Receiver		I_{MAX} bits/sec
		Antenna	Power	Antenna	Noise	
Low altitude satellite	1000 miles −175 db	Isotropic 0 db	10 watts 10 dbw	60 ft 60 db	1000°K −198 dbw	3×10^9 95 db
Communication satellite (low altitude, 2500 miles)	5000 miles −189 db	Isotropic 0 db	1 watt 0 dbw	60 ft 60 db	20°K (maser) −215 dbw	6×10^8 88 db
Communication satellite (stationary, 22,000 miles)	25,000 miles −203 db	Isotropic 0 db	1 watt 0 dbw	60 ft 60 db	20°K (maser) −215 dbw	2.5×10^7 74 db
Same, with directed satellite antenna*	,, ,,	20° beam (19 db) 1° beam (45 db)	,, ,,	,, ,,	,, ,,	93 db 119 db
Lunar probe	250,000 miles −223 db	Isotropic 0 db	1 watt 0 dbw	85 ft 63 db	100°K (param) −208 dbw	10^5 50 db
Mars or Venus probe	2.5×10^8 miles −283 db	Isotropic 0 db	100 watts 20 dbw	85 ft 63 db	100°K (param) −208 dbw	10 10 db
Same, with later improved receiver	,,	,,	100 watts 20 dbw	600 ft 80 db	20°K (maser) −215 dbw	2500 34 db

*The 20° beam will cover the surface of the earth below the satellite so that one satellite antenna will communicate with several ground stations; the 1° beam would be directed to a single ground station, and represents the best current satellite stabilization accuracy.

been used for long range space probe communications, and a FB/FM system is used in the NASA-Bell passive Echo satellite, with ± 30 kc deviation, to give a 20 db improvement.

The data links of current low altitude satellites, forming the first item of Table 2, have frequently merely adopted conventional missile telemetering techniques, in which simple FM is used as the modulation method. It is evident that there is ample margin for such extravagant techniques with present satellite equipment. The communication satellites are in a more difficult position since it is envisaged that future communication loads will require a very large number of voice and TV circuits so that the total information bandwidth may be as high as 20 mc/sec. Furthermore, commercial standards of quality for such signals require high signal-to-noise ratios. For this application, FB/FM systems are currently being advocated.

Even with the best modulation scheme, Table 2 demonstrates clearly that transmission over interplanetary distances must be limited to only a few bits of information per second, at least until the big antennas now under construction are in full operation. Of course, for transmission from space on an interplanetary journey, there will in general be no serious requirement for large information rates, since the journey will occupy periods of months or even years during which conditions are not changing appreciably. The more serious problem occurs at the end of such a journey, when the target planet is being approached. It will then be necessary to expend considerable ingenuity on methods of utilizing the relatively small information rate to give us maximum data. For pictorial transmission, for example, it may be necessary to include in the vehicle a storage system which can take pictures rapidly at the point of nearest approach, and then retransmit the pictorial data over a much longer period of time. There is also scope for some of the many schemes which have been proposed for compression of pictorial data, utilizing techniques such as black to white coding or differentiation of the information with time. In point of fact, this principle of transmitting only changes in information, and not the steady output, could be utilized more fully on many of our current data links with considerable improvement in quality.

SPACE RELAYS

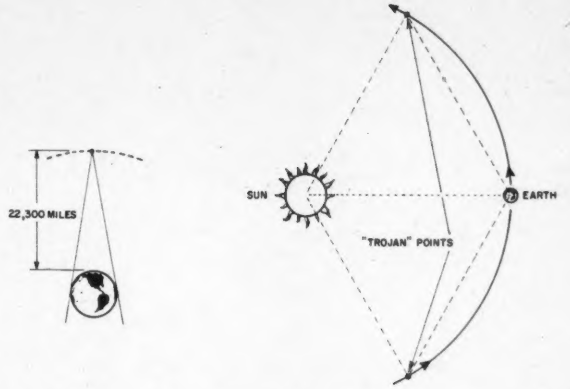
Returning again to a comparison with long distance earth links, a noticeable factor of such links is then the multiplicity of relay stations. We now examine the possibility of corresponding "hilltops" in space, i.e., sites which can see, and are visible from, every point where we may send a space vehicle. Evidently, when we locate the receiving station on earth, or on the moon or another planet, at least half of the field of view is blocked out. In fact, if there is an atmosphere the usable field of view is even less. In addition, space journeys will usually be in the plane of the ecliptic, so that there is a significant probability that the sun will block out the transmission (note its extremely high noise temperature, which presumably extends out into the corona). A space station sufficiently far removed from the earth to have an all-around view, and also to look around the sun, is thus of considerable interest.

There is a further requirement for such stations which has no terrestrial equivalent; we would like to know where they are, without the need for continuous tracking, complex ephemeris computations or large nautical Almanacs. Figure 7 illustrates two possibilities which satisfy these requirements. The first is the well known equatorial "stationary" satellite at a height of about 22,000 miles; the earth only obstructs about

½% of its view of the heavens, and there is no difficulty in finding it. But it does not help appreciably in seeing around the sun, and it is unattractive in cost compared with two or three suitably placed earth stations for all-around viewing. The cost picture changes completely when a stationary satellite with attitude stabilization is already in view as an earth-to-earth communication link. Possibly such future vehicles will carry, in addition to their normal ground-to-ground relay equipment, a ground-to-space relay changing frequencies to the higher values which appear more appropriate for space-to-space links.

The other space relay possibility illustrated in Figure 7 is less well known, involving points in space which we in Lockheed have called the Trojan points. At the two points shown, forming an equilateral triangle with the sun and earth in the plane of the earth's orbit, it is easily shown that the gravitational and centrifugal forces are in equilibrium; the original demonstration of this is attributed to Lagrange. It can further be shown that when the mass of the smaller body is less than 1/25 of that of the larger, the equilibrium is stable. A space vehicle launched with the correct velocity and direction would then be locked in position. An error in initial conditions, or a subsequent perturbation, gives rise to a doubly-periodic motion around the equilibrium point.

The combination of the sun and each planet satisfies this stability requirement without exception, and so does the moon-Earth pair. The strength of the restoring forces, or the permissible error for a given angular deviation, is greatest for the Earth-moon system, followed by sun-Jupiter and then the smaller planets. The sun-Jupiter combination has already captured a few bodies of appreciable size, known as the Trojan satelloids; they were discovered in 1906, more than a century after the theoretical prediction. The Earth-moon and Earth-sun Trojan points are apparently not inhabited by any bodies of visible size at present; they appear to be eminently suitable for space relay stations, particularly for the purpose of looking around the sun.



(a) Stationary satellite **(b) Stable "Trojan" points**

Figure 7
Space relay stations

Needless to say, these special points have also been proposed for other space uses. For example, C. Carruthers (Raytheon) has pointed out their utility as space navigation beacons, and we have also considered the possibility that they might be used as parking places where a "hot" nuclear rocket might be left to lose radioactivity before being reclaimed. It is unlikely, however, that there will be a need for an interplanetary board to allocate uses of these points at present, since there are a reasonable number conveniently located throughout the solar system.

CANADIAN AERONAUTICAL INSTITUTE

NONPROPULSIVE POWER SYSTEMS FOR LONG-TIME SPACE APPLICATIONS†

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SUMMARY

At present, solar photocells are the prominent source of power for long-time, nonpropulsive applications in space, and they will maintain this prominence for powers below a kilowatt or so. Thermoelectric power supplies are being developed for use with solar, radioisotopic and reactor energy sources; these systems will be useful in those applications involving impact or long periods of darkness. Turbogenerator systems employing solar and reactor energy sources are being developed for powers of several kilowatts and up; their use for nonpropulsive applications is several years away. Thermionic emitters are the basis for promising future systems, but their real potential cannot yet be evaluated.

INTRODUCTION

The 10 year space flight program of the National Aeronautics and Space Administration (NASA) calls for launching a total of 260 major satellites and space vehicles in this decade. Anticipated missions include weather observation satellites, communications satellites, satellites and spacecraft devoted to scientific experiments, and manned satellites and spacecraft. It is anticipated that unmanned spacecraft will probe the moon and the near planets and that manned spacecraft will circumnavigate the moon. Manned lunar landings are expected to occur in the 1970's. This 10 year program will, of course, be modified from year to year on the basis of realized progress and the availability of funds.

The purpose of this paper is to discuss briefly the nonpropulsive, long duration, electric power requirements of the NASA space flight program and the power-production systems that are expected to satisfy these requirements. The discussion is intended primarily for those unfamiliar with the field. No attempt will be made to discuss such other topics as:

(a) propulsive power (power production systems for electric propulsion or for attitude control),

(b) short duration power (systems based on chemical energy as the primary energy source), and

(c) power requirements or power-production systems of interest to the US Department of Defense.

The discussion is organized as follows: (1) power requirements, (2) energy sources, and (3) the systems themselves. The systems are divided into three groups: (1) systems in use, (2) systems being developed, and (3) systems just coming out of the laboratory.

POWER REQUIREMENTS

The phrase "power requirements" is a euphemism for a painful compromise. The ultimate desideratum would be, of course, to have unlimited power available. This is not feasible because of the limit on weight that is assigned to the power supply. Consequently, there is always an exchange between the vehicle designer and the payload designer that commences with "How much power do you need?" and "How much power can you give us?", and continues until some compromise is reached that is generally just barely satisfactory to both parties. When future power requirements are described in this paper, it is always in the context of this compromise; that is, an attempt has been made to guess what the compromise will be, not only for space vehicles currently designed but also for those that are only "gleams in their fathers' eyes". Clearly, any such estimate is very uncertain at best.

Long duration electric power requirements can be conveniently considered in three categories: communication power, life support power and instrument power. The significance of the first two categories is fairly obvious from their designations; instrument power is just a handy term for the catch-all category including everything not in the first two categories, such as power for scientific instruments, navigation and guidance equipment, and so forth.

Communication power requirements depend on a number of factors: the rate at which it is desired to transmit information, the desired signal-to-noise ratio, the noise of the communicating equipment, the antenna gains on earth and on the vehicle, the distance, and so forth. The possible range of each of these factors is very great and accordingly the communication power requirements can vary greatly. Let us consider some of these factors individually.

In communication, the rate of information transfer is usually measured in bits per second (bit is an abbreviation for binary digit). Communication power

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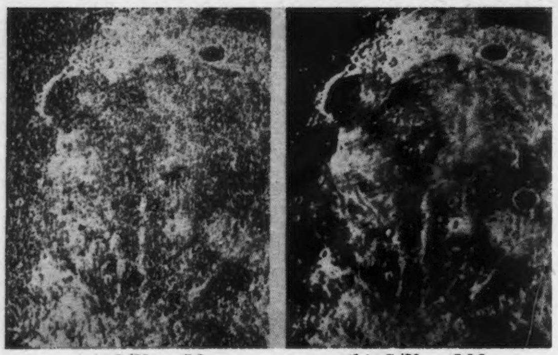


(a) S/N = 10 (b) S/N = 100

Figure 1

Effect of signal-to-noise ratio on lunar photograph

required varies linearly with information rate. The scientific information gathered in space can usually be expressed by a few numbers. A number with 3 significant figures in the decimal system requires about 10 bits for its expression in a binary system. Consequently, the bit rate required for most scientific experiments is low. For example, cosmic ray experiments require information rates of 2 to 4 bits per second; measurement of magnetic fields in space would require about the same information rate; a mass spectrograph might require 20 bits per second; and measurement of atmospheric composition of Mars or Venus from an entering spacecraft might require 100 bits per second.

On the other hand, observations based on sound and sight contain a large amount of information. Even blurry, indistinct photographs contain of the order of 50,000 bits of information, and a clear, well defined photograph may contain 750,000 bits of information. A clear voice transmits of the order of 50,000 bits of information per second. The transmission rate required for information of sight and sound depends on how long a time is available for its transmission. True-time, clear television requires about 10^7 bits per second. However, a 200,000 bit picture can be transmitted in about 33 minutes at 100 bits per second.

For typical scientific satellites of the near future, average information rates of the order of perhaps 100 bits per second are satisfactory. For manned flight, true-time voice communication is desirable, and bit rates of the order of 50,000 bits per second are desirable. The meteorological satellite program includes transmission of many pictures in reasonable periods of time (Tiros I transmitted about 23,000 pictures in a few months), and high information rates are desirable.

The clarity with which information arrives depends on the signal-to-noise ratio. For transmission of a number (e.g., an instrument reading) that can be repeated a number of times, signal-to-noise ratios as low as 2 or 1.5 are acceptable. In picture transmission, higher signal-to-noise ratios are required. Figure 1, taken from Hamilton and Laue[1], shows lunar photographs with signal-to-noise ratios of 10 and 100.

The noise level of communication equipment is being continuously reduced. A good receiver might have a noise power at 1000 megacycles, about equivalent to thermal noise at room temperature. The anticipated introduction of maser techniques might reduce the noise by an order of magnitude.

Antenna gains both on the ground and in the spacecraft can and undoubtedly will be increased. The spacecraft antenna gain can be considerably increased by use of beamed antennas in place of the isotropic antenna, typical of the early space vehicles. Increased antenna diameter both in the spacecraft and on the ground will also increase gain. Combined antenna gains may increase anywhere from 10 to 1000 times, depending primarily on the size of spacecraft antenna that can be carried and the accuracy with which it can be aimed.

The power required for communication varies directly with the square of distance, and the distance can vary from 200 miles for a low altitude satellite directly overhead to the order of 100 million miles for a spacecraft arriving at Mars.

It is clear from the discussion of individual factors that the power required for any particular communication purpose is not fixed but depends on factors under the control of the spacecraft designer (such as spacecraft antenna diameter and information transmission rate) and on factors not under the control of the spacecraft designer (such as ground antenna gains and performance of low noise receivers). For most missions in the NASA 10 year program, the average communication power requirements will not exceed a few hundred watts to a kilowatt. A circumstance that might result in appreciably higher power levels for communication would be the addition to the 10 year program of a mission requiring true-time television transmission. No such transmission is required in the 10 year program at present.

The amount of power required for life support also varies with number of men and length of flight. Various estimates indicate that a power of the order of 100 to 1000 watts per man might be a typical value for the manned flights of this decade.

The amount of instrument power required is generally not large. Typical values for scientific instruments are: magnetometer, 5 watts; micrometeoroid microphone, 2.5 watts; array of six cosmic ray counters, 2 watts; and mass spectrograph, 17 watts. Requirements for navigation and guidance equipment are generally of the same order of magnitude.

In summary, these requirements for given missions result in a broad range of power requirements. Table 1, taken from Slone and Lieblein[2], indicates the level of power available in some of the satellites and spacecraft launched up to April 1960. Future requirements will vary greatly. Most unmanned missions will require powers under a kilowatt. On the low side, the power requirements for passive deflectors for the communications satellite program are of the order of a few milliwatts (Echo I has a 10 milliwatt power supply). Manned missions, on the other hand, will require average powers ranging from the 260 watts of Project Mercury to the order of perhaps 1 to 5 kilowatts. If true-time television transmission becomes an objective, then even higher powers may become desirable; however, almost all missions will fall in the power range of a few kilowatts or less.

TABLE 1.
SUCCESSFUL SATELLITE AND SPACE PROBE LAUNCHES

Name	Origin	Launch date	Lifetime or end	Payload	Power requirements	Power supply	Remarks or experiments
Sputnik I	USSR	October 4, 1957	January 4, 1958	184 lb 83.5 kg		Chem. batt.	Recorded internal temperatures and pressures
Sputnik II	USSR	November 3, 1957	April 13, 1958	1120 lb 508 kg		Chem. batt.	Cosmic rays; solar ultra-violet x-radiation; test animal (Dog)
Explorer I	USA	January 31, 1958	3 to 5 years	10.63 lb 4.82 kg	60 mw	Hg batt.	Discovered Van Allen radiation belt
Vanguard I	USA	March 17, 1958	2,000 years	1.06 lb 0.48 kg		Hg batt. solar cells	Tested solar batteries; revealed pear-shaped earth
Explorer III	USA	March 26, 1958	June 27, 1958	10.83 lb 4.91 kg	55 mw	Hg batt.	Cosmic ray intensity; temperatures; micrometeorite data
Sputnik III	USSR	May 15, 1958	April 6, 1960	2134 lb 968 kg		Chem. batt. solar cells	Analyzed cosmic radiation, composition of atmosphere, etc.
Explorer IV	USA	July 26, 1958	1 year	18.26 lb 8.28 kg	30 mw	Hg batt.	Measured corpuscular radiation
Pioneer I	USA	October 11, 1958	43 hours	85 lb 37.5 kg		Chem. batt.	Density of micrometeorites; measured interplanetary magnetic field
Pioneer III	USA	December 6, 1958	38 hours	13 lb 5.5 kg	180 mw	Hg batt.	Discovered second radiation belt around earth
Project Score	USA	December 18, 1958	January 21, 1959	150 lb 68 kg		Chem. batt.	Beamed human voice from space; messages to and from ground station
Mechta	USSR	January 2, 1959	—	800 lb. 366 kg	30 w	—	First to reach vicinity of moon
Vanguard II	USA	February 17, 1959	200 years	21.5 lb 9.76 kg.		Chem. batt.	Cloud cover satellite
Pioneer IV	USA	March 3, 1959	—	13.4 lb 6.08 kg	180 mw	Hg batt.	Earth-moon trajectory
Explorer VI	USA	August 7, 1959	1 year	142 lb 64.4 kg	11 w continuous 51 w max	Ni-Cd batt. solar cells	Radiation belt; magnetic field; micrometeorite; radio propagation
Lunar probe	USSR	September 12, 1959	Impact on moon, September 13	858 lb 389 kg		—	Impact on moon
Vanguard III	USA	September 18, 1959	30 to 40 years	50 lb 22.7 kg	80 mw	Ag-Zn batt.	Measured magnetic field; intensity of solar x-rays
Lunar probe	USSR	October 4, 1959	—	614 lb 278 kg		Chem. batt. solar cells	First picture of far side of moon
Explorer VII	USA	October 13, 1959	20 years	92 lb 41.7 kg		Hg, Ni-Cd. solar cells	Studied direct solar radiation
Pioneer V	USA	March 11, 1960	—	94.8 lb 43 kg	5w continuous 150 w max	Ni-Cd batt. solar cells	Radio transmission test; radiation belt; magnetic field; micrometeorites
Tiros	USA	April 1, 1960	90 days	270 lb 122.5 kg	18 w	Ni-Cd batt. solar cells	Photographs of cloud cover, meteorological satellite
Transit I	USA	April 13, 1960	16 months	270 lb 122.5 kg	10 w 20 w max	Ni-Cd, Ag-Zn solar cells	Navigational-aid satellite

ENERGY SOURCES

Chemical energy sources are inadequate for long duration space power systems, because of the large weight of chemicals required and, accordingly, only solar and nuclear energy sources are considered herein; systems using chemical energy are discussed in Reference 3, for example. Two types of nuclear energy source are of current interest: radioisotopes and reactors. These three types of energy source (solar, radioisotopic and reactor) will be considered in turn.

Solar energy

The sun is continually radiating energy at a rate of about 0.38 million, billion, billion kilowatts. Extrapolating from considerable past experience, we consider it a highly reliable energy source. Solar energy has the advantages that it does not increase the weight of shielding required for radiation protection, and it creates no nuclear hazards at the launching sites. Solar energy has, indeed, so many desirable characteristics that it is by a considerable margin the most important energy source for nonpropulsive power. There is, however, probably nothing either in this world or out that is an unmitigated blessing, and the sun as an energy source has two drawbacks. It is unfortunately at some distance from us, and the solar energy receptor must be able to see the sun. Let us consider the consequences of these two disadvantages in turn.

The mean distance of the earth from the sun is 92.9 million miles. At this large distance the solar constant (outside the earth's atmosphere, the solar power incident upon a unit area normal to the sun's rays) is about 130 watts/sq ft. Since typical energy conversion devices of interest have efficiencies of the order of 5% to 15%, the energy receptor for a solar power system must have a frontal area of 50 to 150 sq ft/kw. This is an acceptable area for almost all nonpropulsive power systems, those below, say, a few kilowatts. It commences to be quite a design problem in the kilowatt range, however, and for power requirements above a few kilowatts this large receptor area can become a serious drawback. For example, propulsive power requirements range from 30 kw to several megawatts. The frontal area required for a 2 megawatt system is of the order of 100,000 sq ft.

The requirement that the receptor of a solar power system always see and, indeed, point at the sun causes problems. These problems fall into two categories:

(1) orienting the receptor when the vehicle is in sunlight, and

(2) providing power when the vehicle is not in sunlight.

First consider receptor orientation. If the power level is small enough, this problem can be solved by providing fixed receptors variously oriented so that, no matter what the vehicle's attitude, the correct area is intercepted. This approach, however, increases the area of the receptors themselves by a factor of 5 and is therefore practical only for powers at the lower end of the sub-kilowatt range. For higher powers, some electromechanical means is required for orienting the receptor; although the orienting mechanism is fairly low in weight, it does reduce reliability and increase the difficulties of vehicle integration.

The second problem, that of energy storage for periods when the spacecraft is not in sunlight, is a more serious one. If the total accumulated time in darkness is less than a week or two, some chemical energy based, power production system can be used during the periods of darkness but if the accumulated time is longer some method of storing solar energy is required. Solar energy can be stored in a number of forms. It can be stored in the form of heat; it can be converted to electricity and stored in a storage battery; it can be converted to electricity, the electricity used to decompose some compound, and the decomposition products stored; and so forth. Currently, only two methods are of major interest, heat storage and storage batteries.

Heat storage implies that the solar energy conversion device is a heat engine, and therefore thermodynamic considerations require the addition of heat within a relatively narrow range of temperature. Accordingly, substances are required for heat sinks that have large changes in internal energy over small temperature differences, and this generally means that substances with melting points in the temperature range of interest and with high latent heats of fusion are sought. Melting rather than boiling is usually of interest because of the high specific volume of vapours. One substance of considerable current interest is lithium hydride, which melts at about 1250°F and has a heat fusion of the order of 1200 Btu/lb. For a heat engine 10% efficient, about 30 lb of lithium hydride are required per kilowatt hour of electrical energy produced in the dark.

Use of storage batteries generally involves a greater weight than use of heat storage for those missions in which the dark period recurs many times. If charge-discharge cycling hundreds or thousands of times is required, battery weight rises to the order of several hundred to a thousand pounds per kilowatt hour. Nickel-cadmium batteries are normally used for this service. If only a few charge-discharge cycles are required, then batteries of zinc-silver oxide cells can weigh as low as the order of 20 lb/kw hour.

In summary, the sun is a very attractive energy source for nonpropulsive power generation. Its single most serious problem in the sub-kilowatt range is energy storage. In the kilowatt range, the receptor size becomes a problem that increases with increasing power level.

Radioisotopes

Radioisotopic energy sources are of considerable interest for powers in the sub-kilowatt range. Powers are limited to the sub-kilowatt range by several factors, among them that a nuclear reactor is the lighter energy source for powers above a few kilowatts. However, the two dominant factors are availability and hazards. The availability of appropriate radioisotopes is limited, and the launching hazard increases with the power level. Indeed, nuclear hazards constitute a major problem for both radioisotopes and reactor energy sources, and represent one of the principal advantages of solar energy systems.

TABLE 2.
CHARACTERISTICS OF RADIOISOTOPES

Radioisotope	Half life	Curies / Thermal kw	Pounds / Thermal kw	Source
Ce 144	285 days	130,000	0.09	Fission product
Pm 147	2.6 years	2,700,000	6	Fission product
Sr 90	28 years	150,000	2	Fission product
Po 210	138 days	32,000	0.02	Neutron irradiation of bismuth
Pu 238	90 years	31,000	4	Neutron irradiation of uranium
Cm 242	162 days	28,000	0.02	Neutron irradiation of uranium

The power available from radioisotopes continually decreases with operating time, and radioisotopes of long half life are sought. Table 2 lists some isotopes of interest. The alpha emitters (Po 210, Pu 238, Cm 242) have the advantage that they require very little shielding and this simplifies ground handling and keeps weight down. However, these alpha emitters are considerably more hazardous than the others in Table 2 if accidentally released into the atmosphere.

In summary, radioisotopic sources are of interest for power levels below a few hundred watts, particularly for situations in which solar systems have difficulty.

Reactors

Reactor energy sources are handicapped for application in the power range below a few kilowatts by the fact that there is a minimum critical size for reactors. Reactor weights of the order of a couple of hundred pounds are typical of small reactors designed for these applications. This one fact alone limits the range of application to systems of a few hundred watts or greater. In addition, for many applications, radiation shielding is required and this further increases system weight. Because of this, reactors will very likely not be used for nonpropulsive power on any manned missions of this decade. However, there are some missions, even in the sub-kilowatt range for which solar energy systems are unattractive (e.g., Venus soft landings) and for which the power required may be sufficiently high (a few hundred watts or more) for a reactor system to be of interest. As the

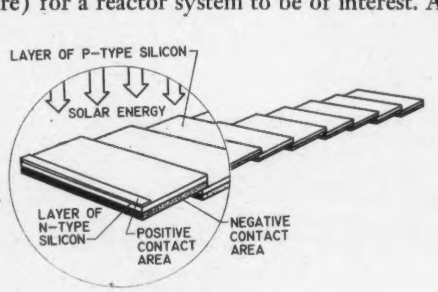


Figure 2
Solar photocells in shingle array

power level rises above a few kilowatts and the receptor of the solar energy system becomes larger, the reactor still weighs a few hundred pounds even up to thermal powers of the order of megawatts. Thus, for high powers (typical of electrical propulsion) reactors come into their own.

SYSTEMS CURRENTLY IN USE

The only long duration power production system currently in use employs solar photocells, usually in combination with rechargeable batteries in order to provide power during periods of darkness. A typical shingled assembly of photocells is shown in Figure 2. A solar photocell consists of a slab of n-type silicon (silicon to which an impurity has been added in order to produce a high concentration of conduction electrons) on whose upper surface a thin layer of p-type silicon has been produced by diffusion of boron into this surface. This p-type silicon has a low concentration of conduction electrons. If the energy of the solar photons absorbed by the cell is sufficiently great to break the covalent bonds holding the electrons to the silicon atoms, electron-hole pairs will be created. The energy required for production of electron-hole pairs is called the "band gap", and it is the minimum energy required to raise an electron from the nonconducting "valence band" to the "conduction band", in which the electron is free to be part of an electric current. If the electron-hole pair is produced in the vicinity of a p-n junction, the electron and hole are separated by the junction, and useful power is produced by the cell.

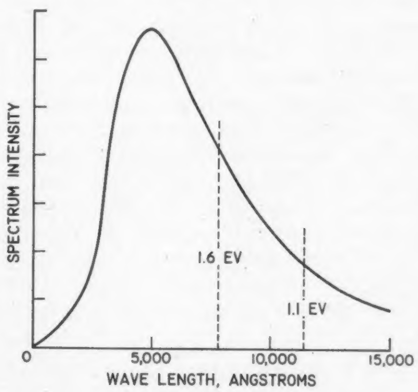


Figure 3
Solar spectrum intensity

Solar photons having energies different from the band gap result in energy losses. Absorbed photons insufficiently energetic to produce electron-hole pairs simply produce heat. Photons whose energy is greater than the band gap produce heat with their excess energy.

Intensity of the solar spectrum is shown in Figure 3; indicated in the figure is the maximum wavelength of photons that will produce an electron-hole pair in silicon. This wavelength corresponds to a photon energy just equal to the silicon band gap, which is 1.1 electron volts. If every incident photon which could produce an electron-hole pair did pro-

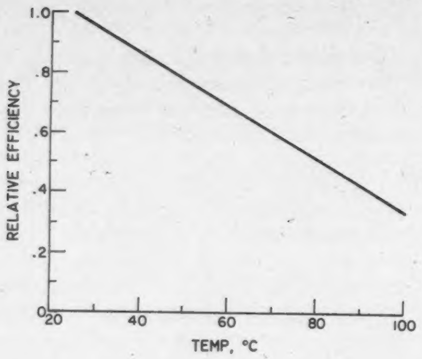


Figure 4
Temperature effects on photocell efficiency

duce one, and if every electron-hole pair was separated by the junction, then the maximum theoretical efficiency would be attained (about 21% or 22% for silicon). Materials other than silicon having different band gaps could be selected. It turns out that the optimum band gap is about 1.6 electron volts, compared with the 1.1 electron volts for silicon, and the maximum theoretical efficiency for this case is about 25%.

The actual efficiency of the silicon photocell is about 10% at a temperature of 25°C. Somewhat higher efficiencies have been attained in the laboratory. The difference between theoretical and actual efficiency comes principally from three sources. First, silicon has a reflectivity of about 30%; surface coatings, in addition to serving other functions, reduce this reflectivity to about 5%. The other two sources of efficiency reduction are the recombination of electron-hole pairs before they are separated by the p-n junction and the photocell's electrical resistance, which is primarily attributable to the thin p-layer. Both the rate of recombination and the resistance of the p-layer are strongly influenced by imperfections in the crystal structure of the silicon. To a large measure, losses, excepting reflection, can be laid to our present inability to manufacture perfect silicon crystals.

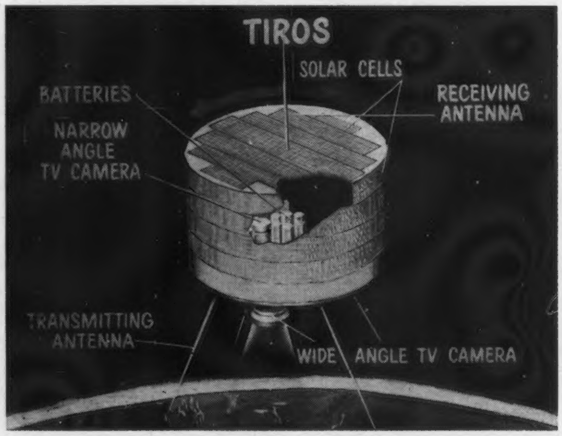
The rate of recombination of electron-hole pairs and the resistivity of the p-layer both increase with increasing temperature. Efficiency, consequently, falls off with temperature as shown in Figure 4. The output of a typical photocell at 80°C is only about 55% of its output at 25°C. This fact affects design of photocell arrays. For example, perhaps the main purpose of photocell coatings is to help keep the photocells cool by increasing the emissivity in the infra-red region around 10 micron wavelength, silicon having an emissivity of only 0.3 to 0.5 in this region. Or again, panel-mounted photocells such as those in Explorer VI (Figure 5(a)) run cooler than body-mounted cells such as those in Tiros I (Figure 5(b)).

The weight of a complete solar cell assembly including structure runs about 100 lb/kw for a perfectly oriented array of photocells 10% efficient operating at 25°C.

Energy storage is a considerable problem with solar photocells. Currently, storage batteries are used, and photocell capacity is increased to permit charging of the batteries while in sunlight. As discussed previously, the number of charge-discharge cycles that the batteries must go through has a strong effect on battery weight. For many missions, a large number of charge-discharge cycles are contemplated (e.g., a near-earth satellite may require 2600 charge-discharge cycles in a 6 month period), and nickel-cadmium batteries are used with only 10% to 15% of the stored energy drained per cycle; this results in a weight of about ½ lb to 1 lb per watt-hour of capacity. Thus, the weight of an unoriented 100 watt photocell system with no energy storage might be of the order of 60 lb; a photocell system to provide this same power at all times in a near-earth satellite might weigh of the order of 150 lb. In addition to increasing weight, batteries decrease reliability because their range of operating temperature is restricted.

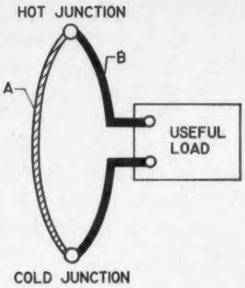
One last factor of importance should be mentioned with regard to solar photocells, and this is their cost. A single 1 cm × 2 cm photocell 10% efficient costs

(a) panel-mounted



(b) body-mounted

Figure 5
Spacecraft with photocells



$S = \frac{E}{\Delta T}$ = **Seebeck coefficient**

ρ = **Electrical resistivity**

K = **Thermal conductivity**

$Z = \frac{S^2}{\rho K}$ = **Figure of merit**

E = **Generated voltage**

ΔT = **Temperature difference**

Figure 6
Thermoelectric device

about $10. The cost for cells alone, therefore, is of the order of $5,000/sq ft, and this area, under ideal conditions, will generate 13 watts of power. When one considers that perhaps 4 packages are required on the ground for one in flight, so that, for example, a mission requiring 50 watts of photocell output from unoriented cells would require about $400,000 worth of cells for the 4 sets, it can be seen that cost is indeed a factor of importance.

SYSTEMS BEING DEVELOPED

Thermoelectric system

Although Seebeck discovered the thermoelectric effect over a century ago, only recently has the new knowledge of material characteristics and the need for new methods for energy conversion resulted in great interest in production of electric power by thermoelectric means. Thermoelectric systems operate on the familiar principle of thermocouples, a principle illustrated in Figure 6. If two dissimilar metals A and B are joined to form an electrical circuit, and if the junctions of materials A and B are maintained at different temperatures, an electrical potential called the Seebeck electromotive force (emf) is produced. This voltage, or potential, could be applied across an electrical load in order to perform useful work.

Heat must be supplied to the hot junction and removed from the cold junction for three reasons. First, a thermoelectric device employs a temperature difference to produce useful work by means of the Seebeck emf and is thus a thermodynamic machine requiring addition of heat at high temperature and rejection of heat at low temperature; high voltage output for a given temperature difference is desired, and this requires a high value for the Seebeck coefficient (Figure 6). Second, the materials of the electrical circuit conduct heat from the hot junction to the cold junction and thereby increase both the heat addition and heat rejection; this loss is reduced if the thermoelectric materials have low thermal conductivity. Third, electrical resistance of the circuit results in heating of the conductors; low electric resistivity is thus desired. The parameter defining the desirable combinations of these three characteristics is the figure of merit Z (Figure 6), for which high values are desired.

The materials having the greatest values for figure of merit Z are semiconductors, a p-type of semiconductor being selected for one leg of the circuit and an n-type for the other leg. Figures of merit of the order of $10^{-3}\ K^{-1}$ are representative of current practice[4, 5]. A power supply that uses the thermoelectric principle for conversion of heat from a radioisotope into 5 watts of electric power has already been built[6] and is called SNAP 3; this power supply weighs 5 lb and has a thermal efficiency of 5%. SNAP 1A[7] is a similar type of device, being developed to produce 125 watts and weighing 175 lb.

SNAP 10 is a thermoelectric power supply presently under development that employs a reactor as its primary energy source[8]. Its weight without shielding is 355 lb, and its design power is 300 watts. Thermoelectric devices employing solar heat lag behind the radioisotope and reactor units; they are either being designed or are in the early phases of development. With present knowledge, solar heated thermoelectric power supplies appear to be both heavier and less efficient than solar photocells.

The basic problems in advancing thermoelectric power supplies for use in space are associated with the thermoelectric materials. Present programs are directed toward development of materials having increased figures of merit or the ability to withstand high temperature, say above 1500°F.

Turbogenerator system

The turbogenerator conversion system is a very familiar one. It is the ground power station taken to flight. There are several differences, of course, in the space turbogenerator system. First, one of the long duration energy sources previously discussed is used. Second, and most important, the system is optimized on the basis of weight rather than cost. These differences produce a number of very significant changes. First, since all waste heat must be radiated directly to space, the condenser temperature is high (of the order of 75% of boiling absolute temperature). Because higher temperatures are required than are provided by steam systems, mercury is currently used as the working fluid. Also, equipment such as regenerators is not used because the increased weight more than offsets the increased efficiency. This optimization on the basis of weight results in an efficiency considerably lower than for ground units, the efficiency running about 10% to 15% for the large systems for propulsion application and even less, perhaps 5% to 8%, for the systems of a few kilowatts and below. This decrease in efficiency with size is primarily a result of low component efficiency for small components. Another important characteristic of space turbogenerator systems is that the systems are hermetically sealed, since replacing lost working fluid results in an unacceptably severe weight penalty. The systems also have bearings that are lubricated by the working fluid.

Four turbogenerator systems for space application are developed or under development. All use mercury as the working fluid. All have turbine inlet temperatures of 1150° to 1300°F and condensing temperatures of 500° to 700°F. The smallest of these is SNAP 1, a radioisotope powered system delivering 470 watts of

electric power[9]. The conversion system for SNAP 1 has been operated on the ground for 2500 hours.

SNAP 2 is a reactor turbogenerator system delivering 3 kw with a conversion weight of the order of 300 lb to 400 lb[10]. The rotating subsystem of SNAP 2 has been operated for periods of hundreds of hours, but the complete system has not been operated.

Two turbogenerator systems whose development is just starting are Sunflower 1 and SNAP 8. Sunflower 1 is a 3 kw system using solar energy. Its weight might be of the order of 650 lb including solar collector and energy storage. The collector alone might weigh of the order of 185 lb. SNAP 8 is a reactor powered system to deliver 30 kw; the conversion system might weigh of the order of 1000 lb to 1200 lb. Two conversion systems will be operated with one reactor to make a 60 kw system. It is intended for propulsion application but could be used for nonpropulsive power if powers of this order are required. Both Sunflower 1 and SNAP 8 are being developed for the NASA. Development of SNAP's 1 and 2 is for the Department of Defense.

Two other turbogenerator systems for space application are under study. These are advanced systems of higher temperature level. One is a 15 kw solar system using rubidium as the working fluid and with a turbine inlet temperature of perhaps 1700°F. The other is a 300 kw reactor system using potassium as the working fluid and with a similar turbine inlet temperature. Each of these is a Department of Defense system.

Turbogenerator systems are attractive for powers above a few kilowatts. Below that level, static conversion devices have about the same weight and are not troubled by the extreme problems of reliability that continually trouble a high performance rotating system that must function unattended for very long periods. From a few kilowatts on up, turbogenerators offer weights unrivaled by any of the currently competitive systems, but this power level is above that required by many of the NASA missions of this decade. However, toward the end of this decade and throughout the next, nonpropulsive power requirements will undoubtedly rise, and the utilization of turbogenerator systems of higher power levels will accordingly increase.

SYSTEMS STILL IN THE LABORATORY

There are a number of ideas for power production systems that are still in the laboratory. Time and space do not permit discussion of them all. They share one common quality; it is too early for their performance potential to be evaluated or for the extent to which they will be used in actual flight to be determined. But one system in this large group stands out because, first, it is potentially a low weight system and, second, it is just entering the development phase. This system is based on the thermionic emitter. The thermionic emitter (Figure 7) consists of a hot plate (cathode) and a cold plate (anode) separated either by a vacuum or by a low density, ionized vapour. The temperature of the hot plate is raised to a level sufficiently high to literally boil electrons out of the cathode; these

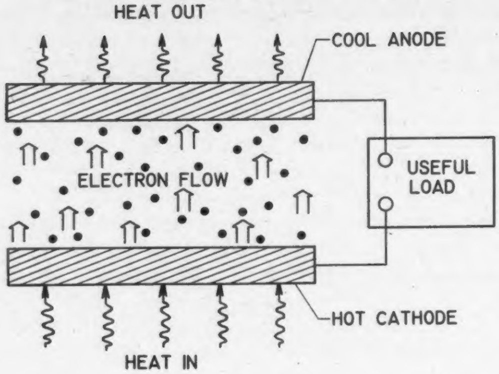


Figure 7
Thermionic emitter

flow to the anode and pass through the load back to the cathode, thereby delivering power. Since the thermionic emitter converts heat to electricity, it can use any of the energy sources discussed previously, solar, radioisotope or reactor.

The basic principle of the vacuum thermionic emitter has been known for a number of years. Two factors deterred its earlier development. First, high temperatures, say above 2000°F, are required to provide sufficient electron emission from the cathode. Second, the electrons flowing between the cathode and the anode create a space charge that tends to reduce electron flow and thereby to reduce power output. The high temperature problem has not been ameliorated, but over the years since the first discovery of thermionic emission, technology has so progressed that these high temperatures might soon be successfully handled in a power system. There are two methods of current interest for reducing or eliminating the problem of the space-charge barrier. The first approach is to reduce the spacing between the cathode and the anode. This reduction in space-charge barrier increases current density inversely with the square of the spacing. Small emitters with spacings as small as 0.0004 inch have been successfully operated for short periods in the laboratory. With this small spacing and with a cathode temperature of about 2300°F efficiencies of 13% and power densities of 0.8 watt/sq cm of cathode area have been attained[11].

Another approach to reducing the space-charge barrier is to introduce cesium vapour into the space between the cathode and the anode. If the cathode temperature is sufficiently high, enough of the cesium vapour becomes ionized for the cesium vapour to behave like a plasma, and it tends to eliminate or reduce the space-charge barrier. This approach to reduction of space charge is relatively new and appreciably increases the current and power density. Unfortunately, the cathode temperature required for ionization of the cesium is very high, and cathode temperatures of perhaps 3000°F or greater are required for high performance plasma filled thermionic emitters. In the laboratory efficiencies of the order of 7% to 13% and power densities of the order of 20 to 30 watts/sq cm have been obtained[12, 13]; unfortunately,

these results have been obtained with cathode temperatures of the order of 4000°F or higher. Considerable work is now in progress to try to obtain this promising performance at lower temperature.

At present, it is not possible to evaluate specific weights that might be obtained from reliable, long duration thermionic emitter systems for space operation. The systems clearly have potential for high performance. On the other hand, they are high temperature systems, and the concessions in performance needed to obtain reliability and endurance for space operation cannot presently be determined.

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EIGHTH ANGLO-AMERICAN CONFERENCE

LONDON, ENGLAND

3rd to 14th September, 1961

Members of the Institute proposing to attend must notify the Secretary, C.A.I., before the 15th July, 1961.

(For further particulars see page 256)

C.A.I. LOG

SECRETARY'S LETTER

ECONOMIES

This letter must be written before the Annual General Meeting takes place but I think I can safely forecast some of the results of that Meeting and of the meeting of the Council held in conjunction with it. The outstanding need at the present time is to balance the budget and, though we economized last year so far as we could without seriously reducing our performance, we still spent too much money. Something has got to be done to carry the Institute through these difficult times, even at the temporary sacrifice of some of our services.

This Journal is by far our heaviest expense; printing costs have gone up and advertising revenue has gone down – rather sharply of late. In April we eliminated some features which had no bearing on the technical content; but that was not enough and now we must reduce the technical content as well. This is a great pity because there is no lack of good material and in future some of it will have to be published elsewhere. We shall have to confine ourselves to the cream. The plan is that instead of four or five papers in each issue we shall have to limit ourselves to two or three. But we shall retain the present monthly frequency and, except that it will be a little thinner, the Journal will look much the same as it has in the past.

This reduction will not only save on printing costs but on the work of this Headquarters, enabling us to consider reducing our present staff. Consequently Mr. W. A. Chisholm, who has been Assistant Secretary and Editor of the Journal for the last two years, will leave us at the end of May; he will be gone by the time this is published (and I can only hope that we shall somehow manage to get it published without him).

I.A.S. STUDENT CONFERENCE

The Detroit Section of the IAS held its Regional Student Conference on the 6th May at the University of Toledo, and this year they kindly invited the CAI to participate. I should explain that, in the first place, the IAS and the CAI use the terms "Sections" and "Branches" in exactly opposite senses and, in the second, each IAS Student Branch is associated exclusively with one university – it is not dependent on a Section, as our Student Sections are dependent on Branches. However the Detroit Section exercises some parental control over all the Student Branches in its area and once a year it organizes a Conference, one of the Sections acting as host.

It was an ambitious and typically hospitable gesture to invite the CAI to add an international flavour to this year's Conference. CAI Students were invited to present papers and, because they were not, strictly speaking, eligible for the IAS Student prizes from the Minta Martin fund, the IAS invented a special international category and raised the prize money from local industry.

With the co-operation of Professor Poppleton of Toronto and Mr. Conrath of Montreal we arranged for two students from each of these Sections to attend; they were Mr. R. C. Radford and Mr. P. R. Stephens from Toronto and Mr. B. R. Thomas and Mr. I. Wygnanski from Montreal. I went along too.

It was a most enjoyable affair. There were about 55 present; the morning and afternoon were devoted to the presentations of some 14 papers, including one by our Mr. Wygnanski, and in the evening we had a semi-formal banquet, at which the judges' awards were announced and the prizes presented. I got a little bewildered at this point and I am not quite clear who won what; I think that there was a tie for first international honours, which was very gratifying.

The Principal Speaker at the banquet was Mr. Charles Tilgner, F.I.A.S., A.F.C.A.I., Vice-President of the IAS, who spoke on "The Importance of the Professional Society".

Our most sincere thanks are due to Mr. W. S. Spreitzer, Chairman of the Detroit Section, who went to enormous trouble to see that we were well looked after, and to our hosts at Toledo, Professor G. L. Heath, Faculty Advisor, and Mr. D. A. Duszynski, President of the Toledo Student Branch.

NATIONAL CONFERENCE ON TECHNOLOGICAL TRAINING

Another event of the month in which the CAI took part was the National Conference on Technological Training originated by the Department of Labour and held in Ottawa on the 9th and 10th May. Professor T. R. Loudon attended as the official delegate of the Institute, and I put in a brief appearance, as an observer, at the plenary sessions. I will not venture to give the details of the decisions reached; we must await the final Report

1961-62 Executive Committee: Halifax-Dartmouth Branch (l to r) Mr. W. G. Stewart, Vice-Chairman; Professor O. Cochkanoff, Chairman; and Mr. V. W. Bowers, Treasurer. Unfortunately this picture does not include LCDR N. L. Brown, Secretary.

of the Conference. However I can say that the purpose of the Conference was to discuss the Technologist level, that is, the level of graduation from the various Provincial Institutes of Technology, and to consider the desirability of establishing some sort of country-wide standard.

DR. THEODORE VON KARMAN

The 11th May was Dr. von Kármán's 80th birthday and the Air Force Office of Scientific Research arranged a Symposium and Banquet in Washington, D.C. to mark the occasion. The ceremony was co-sponsored by some 37 scientific and professional organizations from all over the world, the CAI amongst them. Dr. J. J. Green, our Vice-President Elect, attended the Banquet as our representative. For a birthday present, he gave Dr. von Kármán an excellent little book of photographs of the Canadian scene, suitably inscribed and bearing the Institute's crest. impressed in gold on its cover. I understand that other presents comprised certificates and honours of all sorts, but since Dr. von Kármán is already an Honorary Fellow of the Institute, the highest honour we can bestow, we thought that we should try to encourage him to come to see us more frequently and we hoped that this book would serve that purpose.

BRANCHES

The Branches have been very active lately, holding their Annual General Meetings and installing their new Officers. I will run through their reports in chronological order.

Montreal — 19th April

Following their usual practice the Montreal Branch held a Dinner Meeting on the 19th April. There were 81 members at the dinner and the attendance increased to 138 to hear Mr. F. C. Phillips of Canadair speak on "The Development History of the CL-41". W/C H. J. M. Londeau, the Branch Secretary, was in the Chair and introduced the speaker. He also asked the President, who was present, to say a few words to the Branch.

W/C Londeau sent me an outline of Mr. Phillips' talk and it seems very similar to the paper he presented so ably at the Test Pilots Symposium in November. He is, of course, an expert on the subject of the CL-41 and an excellent speaker. Mr. J. W. J. Truran thanked him on behalf of the Branch at the conclusion of the meeting.

Halifax-Dartmouth — 19th April

The new Officers of the Halifax-Dartmouth Branch were installed at the meeting on the 19th April, held in the Wardroom Mess at HMCS Shearwater. Mr. R. Wallworth, the Past Chairman 1960-61, was in the chair in the absence of the Chairman, LCDR G. M. Cummings, who was at sea. The incoming Executive Committee consists of Professor O. Cochkanoff, Chairman, Mr. W. G. Stewart, Vice-Chairman, Mr. V. W. Bowers, Treasurer, and LCDR N. L. Brown, Secretary. The Fairey Aviation Company has recently presented the Branch with a gavel which was ceremoniously handed over to Professor Cochkanoff as he took office.

With these formalities and the presentation of some new Certificates of Membership out of the way, Mr. Stewart introduced Mr. T. E. Fessenden, Assistant Chief of the Weapons System Section of the Grumman Aircraft Engineering Corporation, who spoke on "Solutions to Aircraft-Stores Separation Problems". Mr. Fessenden had originally tried to visit Halifax-Dartmouth in November but had been unable to get through on that occasion. This time the weather had been a little kinder. Lt A. M. Percy has sent me an excellent report on Mr. Fessenden's talk and I am sorry that I cannot reproduce it here. It seems to have been a most informative paper on an unusual subject. Mr. E. C. Garrard thanked the speaker and observed that he was the first lecturer from the USA to have visited the Branch; he asked for more.

Quebec — 20th April

Following his postponed itinerary, Mr. Fessenden went on to Quebec after leaving Halifax-Dartmouth and spoke to the Branch there on the 20th April. Dr. H. M. McMahon, the Branch Chairman, presided at the meeting, which was held in Laval University with 15 present. (There were 40 present at Halifax-Dartmouth but, in proportion to the size of Branch, Quebec's was the better turn-out.)

I should like to express the Institute's thanks to Mr. Fessenden for visiting these two Branches, after his aborted first attempt, and to Mr. Charles Tilgner of Grumman, who arranged the tour for us.

Edmonton — 21st April

The Annual General Meeting and Dinner Dance of the Edmonton Branch was held on the 21st April in the Isle of Capri Supper Club. (How do they get a name like

Mr. T. E. Fessenden addressing the April meeting of the Halifax-Dartmouth Branch.

that in Edmonton?) There were 87 people there and, according to the report from the Branch Secretary, "it was a very successful meeting and everyone had a wonderful time". At any rate the proceedings opened seriously enough with the presentation of the Branch Annual Report by the retiring Chairman, Mr. R. W. Van Horne, and the introduction of the incoming Executive Committee comprising Mr. J. W. McNee, Chairman, Mr. A. J. Quick, Vice-Chairman, Mr. K. H. Hegstrom, Secretary-Treasurer, and F/L T. L. Hogg, Chairman of the Programmes Committee. (Unlike other Branches they elect their Chairman of the Programmes Committee in Edmonton.)

Winnipeg — 25th April

Because W/C C. J. Evans, the Branch Chairman, has now been spirited away from Winnipeg by the RCAF, Mr. H. W. Grant, the Vice-Chairman, is carrying on for the rest of the season and he presided at the meeting held in the Winnipeg Flying Club on the 25th April. This meeting was attended by 37 members who heard Mr. C. M. Hovey speak on "Modern Developments in Metallurgy and Metal Processing". Mr. Hovey used to teach at the University of Manitoba and, though he is now Test and Development Engineer at Bristol Aero-Industries, it seems that he has lost none of his skill as a lecturer. Mr. Baker, in his Secretary's Report on the meeting, tells me that a manuscript may be available later on and I hope it will be; for evidently this talk was an outstanding survey of the subject, in relatively straightforward terms. It might be suitable for publication in the Journal.

Calgary — 26th April

This again was a Branch Annual General Meeting, held in the RCAF Officers' Mess at Station Lincoln Park, on the 26th April. Mr. J. Stanley, the retiring Vice-Chairman, was in the chair in the absence of Mr. G. H. Fenby, who has moved to Vancouver. There were 28 present.

The Branch Committees presented their several reports and the new Executive Committee was installed. The incoming Officers are Mr. W. R. Burge, Chairman, S/L R. S. Crosby, Vice-Chairman, Mr. C. W. Naigle, Treasurer, and Mr. W. G. Jones, Secretary.

Part of the ceremony consisted of presenting the Branch Student Award to Mr. A. H. Smolkowski of the Southern Alberta Institute of Technology, for his paper entitled "Investigation into Aircraft Drag".

Toronto — 3rd May

The Toronto Branch held its Annual General Meeting at the Four Seasons Motor Hotel on the 3rd May. It took its usual form of a dinner followed by a talk, and this year the talk had little to do with aeronautics. In fact it dealt with mountains; Professor J. T. Wilson of the Department of Geophysics, University of Toronto, entitled his address "The International Geophysical Year" and in the course of tracing the world's mountain ranges, he showed coloured slides of his world tour in the late 1950's. He was introduced to the audience of 75 by Dr. J. H. T. Wade and Mr. J. J. Czarnecki moved a vote of thanks at the end.

Mr. C. H. Bottoms was in the chair and introduced the 1961-62 Executive Committee as follows: Mr. W. T. Heaslip, Chairman, Professor E. D. Poppleton, Vice-Chairman, Mr. R. H. McLachlan, Treasurer, and Mr. R. Parker, Secretary.

At the Montreal Astronautics Group Meeting (l to r) Mr. C. L. Bernier, Branch Treasurer; Mr. D. R. Taylor, Branch Chairman; Mr. K. J. Bossart; Dr. H. J. Luckert, Group Chairman; and Mr. D. Bogdanoff, Past Chairman, Astronautics Section.

Ottawa — 17th May

Ottawa, too, held its Annual General Meeting this month, with the worst attendance this year, 17. This did not constitute a quorum, so that resolutions put to the meeting will have to be voted upon by mail. Even the Chairman, Mr. G. D. Watson, was unable to attend – I know that his excuse was genuine – and G/C E. P. Bridgland, the retiring Vice-Chairman, presided and presented Mr. Watson's Report. He also announced the new Executive Committee as himself, as Chairman, Professor G. S. Glinski, Vice-Chairman, Mr. D. H. E. Cross, Treasurer, and Mr. D. W. Pounder, Secretary.

Then Mr. K. J. Radford gave us a talk which he called "A Review of the Problems of Satellite Rendezvous". This is a comparatively new subject and Mr. Radford handled it very skilfully. The fact that it was a small and intimate meeting coupled with Mr. Radford's straightforward presentation produced a very lively discussion afterwards. A bigger gathering might have spoiled it, but the 300 odd members who stayed away missed one of the best meetings of the season.

SECTIONS

Astronautics: Montreal Group

In collaboration with the Montreal Branch, the Montreal Group of the Astronautics Section held a meeting on the 26th April to hear Mr. K. J. Bossart, Technical Director, Convair Astronautics, speak on "The Atlas Missile and its Potential in Space". It was a dinner meeting attended by 65 members and it ran from 8.00 to 11.00, which is a fair indication of the interest shown and the discussion that took place.

This was a very creditable performance considering that there had been a regular Branch Meeting only a week before.

THE OLD ORDER CHANGETH

With the exception of the Winnipeg and Quebec Branches, I think I now know the names of the 1961-62 Executive Committees across the country; I wish them a good term. We all owe a great deal to the retiring Officers, who have given their time and effort to last year's programme and I am glad to know that many of them have found places on the new Committees and will give us another year of their service.

ANNOUNCEMENTS

TURNBULL FIELD

The new municipal airport building at St. John, N.B. will be formally opened on the 20th June, 1961, and, at the same time, the airport will be named "Turnbull Field", in memory of Dr. W. Rupert Turnbull, who was born in St. John.

The suggestion that the airport should be named after Dr. Turnbull originated with the Institute in 1957 – as a part of the nationwide celebrations of the Fiftieth Anniversary of Powered Flight in Canada in 1959 – but it was turned down at that time. It has since been revived by the local authorities.

A plaque, presented by the Institute, will be unveiled by Mr. Donald Turnbull, Dr. Turnbull's son. The ceremony will start at 11.00 am and it is hoped that as many members as possible will attend.

ANGLO-AMERICAN CONFERENCE

As announced in the last issue the Eighth Anglo-American Conference will be held in London from the 3rd to the 14th September, 1961. The technical sessions will start in the second week, from the 11th to the 14th; during the first week the programme will comprise visits to various plants and facilities, including the S.B.A.C. Show.

Registration

Those planning to attend the Conference must register in advance and should obtain application forms from the Secretary and return them to him before the 15th July. Particulars of the C.A.I. delegation must be transmitted to the Royal Aeronautical Society shortly thereafter.

The Registration Fee of $35 will cover admission to the Conference, the associated paperwork and a copy of the Proceedings.

It would be helpful if those planning to attend would get in touch with the Secretary as soon as possible. Please do not put it off until the last moment.

NEWS OF MEMBERS

I. M. Hamer, F.C.A.I., has been appointed to the Royal Commission on Government Organization as its representative in the Toronto area.

B. S. Shenstone, F.C.A.I., has been named President-Elect of the Royal Aeronautical Society.

W/C H. J. M. Londeau, A.F.C.A.I., formerly Detachment Commander of 1102 TSD at Canadair, has been appointed Commanding Officer of No. 11 TSU Montreal, with effect from the 1st June.

S. H. E. Garside, M.C.A.I., with a group of associates, has formed a company known as Cor-Del (Corporate Development Ltd.) in Toronto for the development of new products, marketing promotion, product analysis etc. Mr. Garside has been elected President of the company.

A. S. Punter, Technical Member, has joined the Tool Requirements Department of Canadian Pratt & Whitney Aircraft Co. Ltd.

ADMISSIONS

The following is a list of admissions and advancement in grade of members during the month of May 1961.

Associate Fellow

H. Haessler
G. F. W. McCaffrey (*from Member*)
A. Rose (*from Member*)
L. W. Smith (*from Member*)

Member

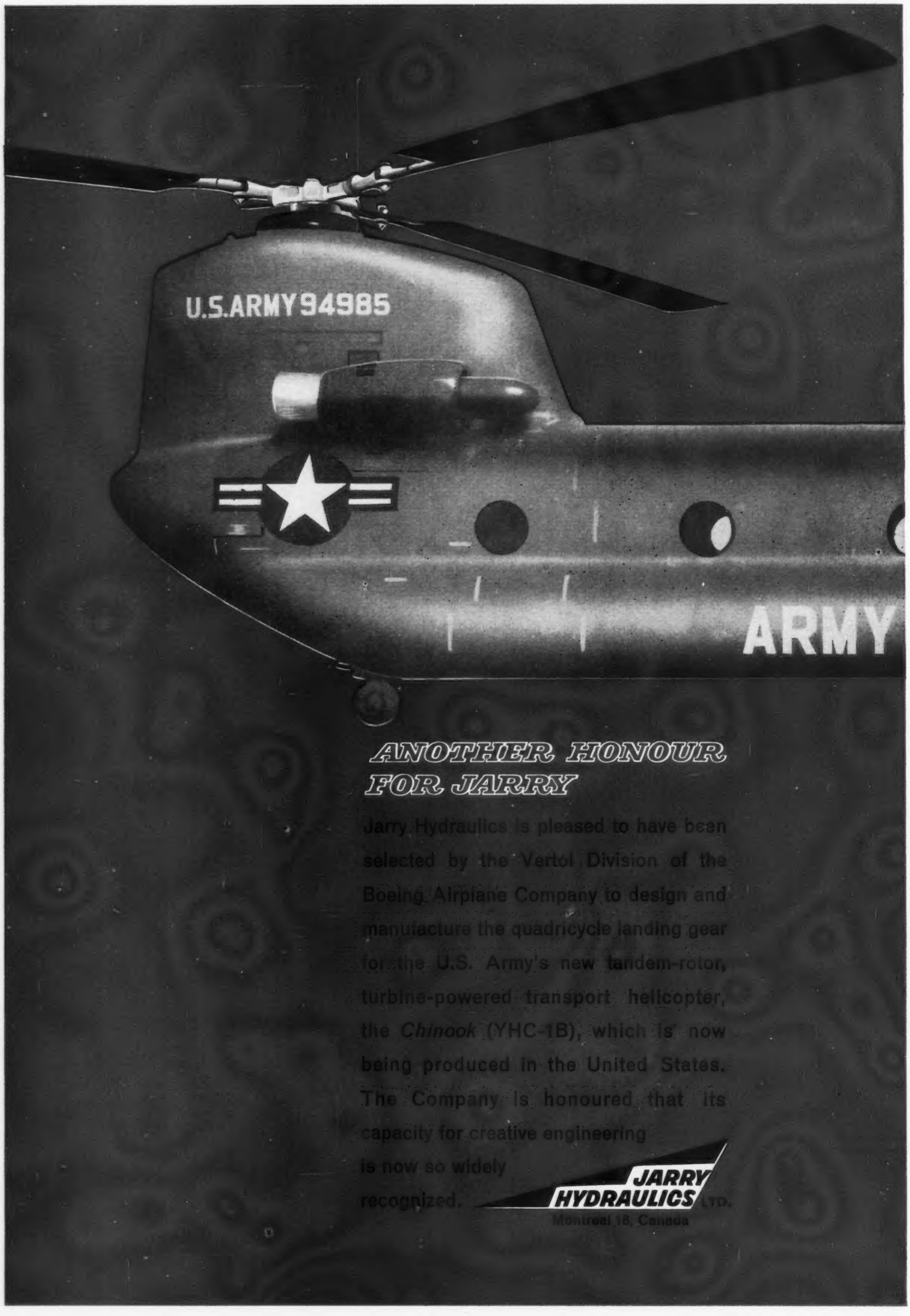
J. A. Charnock
S/L C. C. Cooling
W. J. Dick
H. Dubeta
B. J. Hallam
G. H. Hodgson
J. R. Hook
B. Lang
A. L. Neuburger
F/O P. S. Perry (*from Technical Member*)
H. T. Redick (*from Technical Member*)
J. A. Sagar (*from Technical Member*)
J. A. Sefsik
S. S. Swanson (*from Technical Member*)
J. C. Vrana (*from Technical Member*)
Dr. H. F. Waldron (*from Technical Member*)
A. M. Weber
F/L R. A. White (*from Technical Member*)
W. R. White (*from Technical Member*)

Technical Member

A. Ainsworth
V. T. Baker
C. A. Bates
C. B. Bohlken
L. A. Croucher (*from Junior Member*)
C. M. Cumming
R. Knystautas (*from Student*)
T. W. Letham
F/O R. J. Lawson (*from Student*)
J. R. McLeod (*from Student*)
B. Mills
P. S. Moller (*from Junior Member*)
W. Moody
E. G. Stonier (*from Junior Member*)

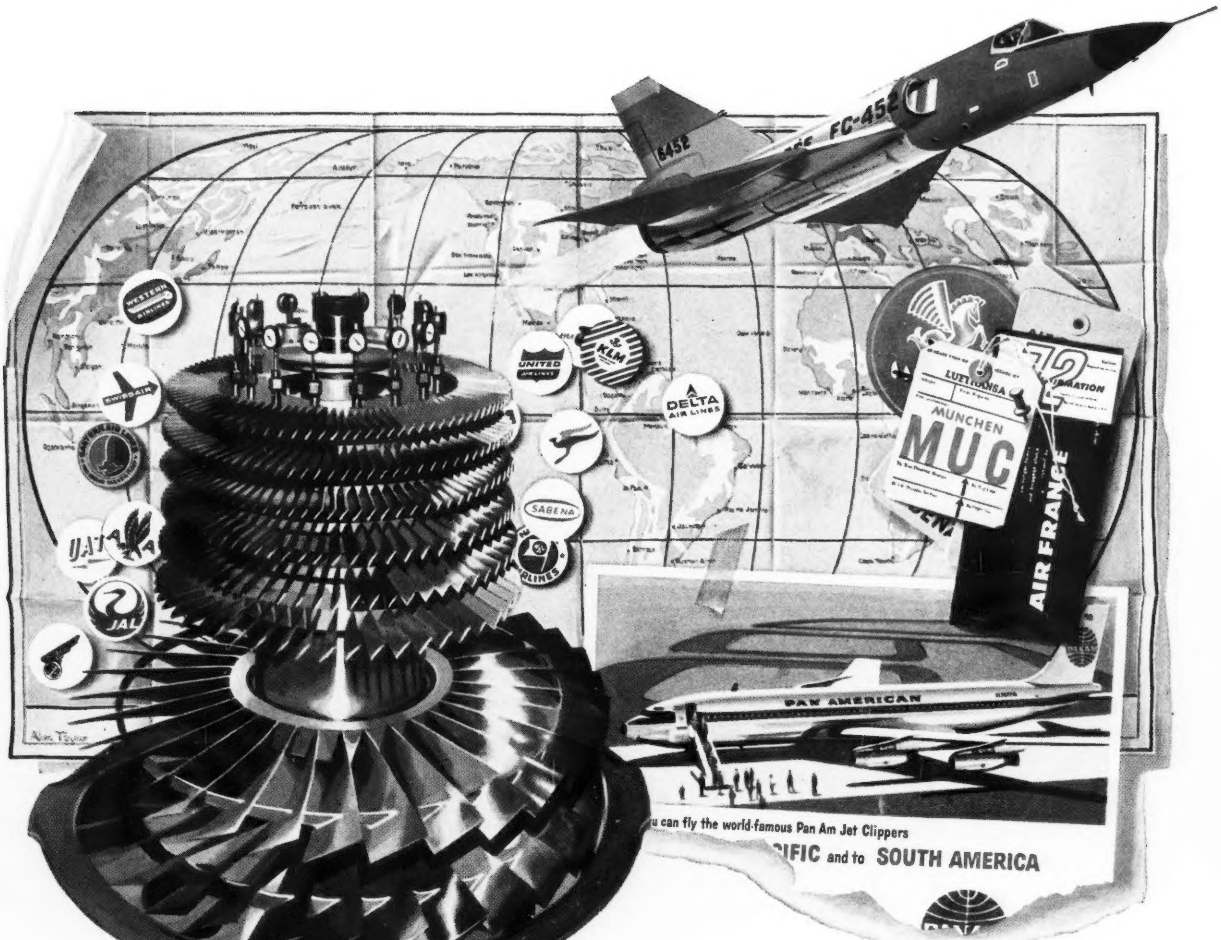
Student

F/C B. W. Ailey
F/C R. O. Brown
F/C T. C. Calow
F/C G. A. Cooper
O/C R. A. Delaney
F/C K. S. Drolet
F/C P. J. Dunne
F/C R. E. Gunn
F/C A. D. Lee
F/C J. W. Palmer
F/C A. L. Paterson
F/C R. E. Prichard
R. C. Radford
F/C E. B. Robertson
F/C R. A. Low
O/C J. M. Saunders
O/C L. J. Staples
F/C H. J. Stevens
O/C R. L. Stickley
F/C D. P. Szabo
O/C D. J. Tretiak
O/C R. E. Uptigrove


U.S.ARMY 94985
ARMY
ANOTHER HONOUR FOR JARRY
Jarry Hydraulics is pleased to have been selected by the Vertol Division of the Boeing Airplane Company to design and manufacture the quadricycle landing gear for the U.S. Army's new tandem-rotor, turbine-powered transport helicopter, the *Chinook* (YHC-1B), which is now being produced in the United States. The Company is honoured that its capacity for creative engineering is now so widely recognized.
JARRY HYDRAULICS LTD.
Montreal 18, Canada

SINCE GEOGRAPHY BECAME JETOGRAPHY continents have shrunk and oceans have dwindled to the compass of a few flying hours. In this significant progress Canadian Pratt & Whitney Aircraft are proud to be associated with the Pratt & Whitney Division of United Aircraft Corporation. P. & W. engines power *more* transport aircraft than those of any other engine maker—as well as the world's *fastest* military and commercial jet aircraft. Soon Canadian Pratt & Whitney will add its new Canadian designed and developed PT6 gas turbine engine to the long line of power plants bearing the world-famous Pratt & Whitney name.

CANADIAN PRATT & WHITNEY AIRCRAFT
COMPANY, LIMITED • LONGUEUIL, MONTREAL, QUE.



A SUBSIDIARY OF UNITED AIRCRAFT CORPORATION

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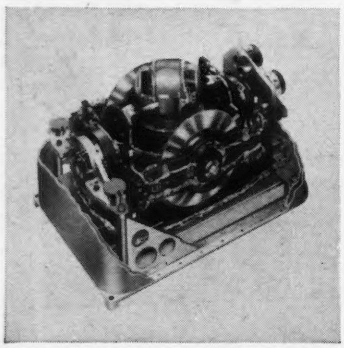
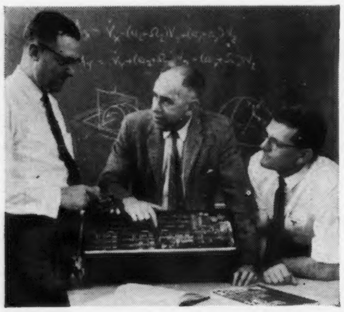
LITTON INERTIAL NAVIGATIONAL SYSTEM ADOPTED BY R.C.A.F.

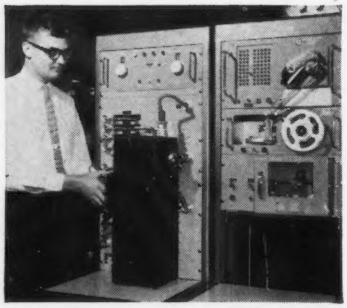
Advanced LN-3 Inertial Platform Features Two Unique, Floated 2°-of-freedom Gyroscopes



The Litton LN-3 Inertial Navigation is a system in being. Production orders for this system, to be used in the RCAF's CF-104 and the NATO F-104, are larger in number than those for any other inertial guidance system. The LN-3, consisting of a two-gyro, four-gimbal platform, computer, adapter, and controls, weighs less than 80 lbs. installed. Smaller systems designed for orbital and sub-orbital guidance, weighing less than half as much, are in development.



To achieve the development of the Litton platform, consideration had to be given to entire new concepts in inertial components, design, packaging, production and testing techniques. Litton developed a unique two-degree-of-freedom gyro measuring three inches in diameter, four inches in length, and weighing only two pounds, that provides a random drift rate capability of less than 0.01 degrees per hour. The two gyroscopes used are packaged in a "dumbbell" configuration which is retained in a four-axis gimbal mechanism. This permits unrestricted angular maneuverability of the vehicle without incurring platform gimbal-lock.



Litton Systems (Canada) Limited is engaged in the manufacture, testing and calibration of LN-3 sub-systems, ground support equipment, etc. If you have experience in fire control, analog, servo or gyro systems, this may be your opportunity to gain further knowledge in this continuously expanding field in Canada with one of the world's leaders in inertial navigation systems.

Opportunities also exist for Field Service Representatives.

To apply, write to the Personnel Manager.

LITTON SYSTEMS
(CANADA) LIMITED
123 Rexdale Blvd., Rexdale, Ontario.

A SUBSIDIARY OF LITTON INDUSTRIES, INC., BEVERLY HILLS, CALIFORNIA.